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XLR129-P-1

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FOREWORD

This Technical Report presents the design and operational characteristics of the XLR129-P-1 reusable rocket engine. In addition, it presents engine parametric performance, size, and weight data for future flight engines that could result from an engineering development program based on this engine concept. This report is issued as a special report in accordance with the requirements of Contract F04611-68-C-0002.

This publication was prepared by the Pratt & Whitney Aircraft Florida Research and Development Center as PWA FR-3108.

Classified information has been extracted from (asterisked) documents listed under References.

This Technical Report has been reviewed and is approved.

Ernie D. Braunschweig
Major, USAF
Program Manager
Air Force Rocket Propulsion Laboratory

UNCLASSIFIED ABSTRACT

This special technical report presents information and data on the XLR129-P-1 rocket engine. Information is presented for both the demonstrator engine and flight engine versions of this rocket engine. A general description and pertinent technical information are presented for the demonstrator engine. The demonstrator engine program schedule is also presented. Parametric design, performance, cost, and schedule data are presented for the flight engine. This technical report has been prepared for the use of airframe manufacturers and government personnel who are conducting mission and vehicle studies.

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ILLUSTRATIONS

ILLUSTRATIONS (Continued)

ILLUSTRATIONS (Continued)

ILLUSTRATIONS (Continued)

TABLES

LIST OF SYMBOLS AND ABBREVIATIONS

ADP	Accessory Drive Pad
F_{alt}	Thrust at Altitude
F_{vac}	Thrust at Vacuum
FVM	Fuel Vent Manifold
g	Gravitational Constant
I_s	Specific Impulse
$I_{s_{alt}}$	Specific Impulse at Altitude
I_{vac}	Specific Impulse at Vacuum
L/D_t	Length/Throat Diameter
MC_s	Maximum Performance Nozzle
MSA	Minimum Surface Area
NPSH	Net Positive Suction Head
OVBD	Overboard
OVM	Oxidizer Vent Manifold
PFRT	Preliminary Flight Rating Test
r	Mixture Ratio
TBO	Time Between Overhaul
TVC	Thrust Vector Control
ϵ	Nozzle Expansion Ratio
ϵ_p	Primary Nozzle Expansion Ratio

SECTION I
INTRODUCTION

SECTION I
INTRODUCTION

(U) This technical report provides a general description, pertinent technical information, and the program schedule for the XLR129-P-1 demonstrator engine being designed, fabricated, and tested under Contract F04611-68-C-0002. In addition, parametric data on the performance, weight, and size of reusable rocket engines which could result from an engineering development program is provided for use in vehicle and mission studies.

(C) Data are presented for high pressure (3000-psia chamber pressure) staged-combustion, pump-fed, oxygen-hydrogen, engines with transpiration-cooled thrust chambers and regeneratively and dump-cooled nozzles.

(U) By combining interchangeable nozzle extensions with a basic turbopump and combustion chamber module, the engine can be tailored to specific rocket stage requirements. Data are provided for conventional fixed nozzle engine configurations and two-position nozzle engine configurations for various nozzle expansion ratios and contours. The two-position nozzle concept is based on part of the nozzle being retracted over the forward portion of the thrust chamber during low altitude operation and extended to the high area ratio position for high altitude operation. This principle enables the high pressure engine to have an exhaust nozzle that is optimized for both high altitude and low altitude operation. In addition, the two-position nozzle will provide high specific impulse in vehicle installations where the length of the engine is limited. Dump cooling is used for the extendible portion of the nozzle, which also permits lightweight nozzle construction.

SECTION II
SUMMARY

SECTION II
SUMMARY

(U) Information presented on the XLR129-P-1 demonstrator engine include: the overall engine design characteristics, an installation drawing, a flow schematic, the engine internal design parameters, the component and engine weight, a schedule of the demonstrator engine program, start and shutdown thrust versus time curves, and engine/vehicle interface propellant requirements.

(U) The parametric performance data, which are presented for the LR129-P-1 flight rocket engine, include delivered vacuum specific impulse versus nozzle expansion area ratio for three nozzle contours and five mixture ratios; specific impulse versus altitude for fixed nozzle and two types of two-position nozzle engine configurations. In addition, parametric data are presented for engine weight, diameter, and length versus nozzle expansion ratio for three nozzle contours. Also included is a throttling curve that presents delivered vacuum specific impulse at various throttling conditions for five different mixture ratios.

SECTION III

DEMONSTRATOR ENGINE

Page

SECTION III
DEMONSTRATOR ENGINE

A. ENGINE CHARACTERISTICS

(U) The design and demonstration characteristics for the XLR129-P-1 demonstrator engine are shown in Table I.

(C)(U) Table I. Demonstrator Engine Characteristics

Nominal Thrust	250,000-lb vacuum thrust with area ratio of 166:1 244,000-lb vacuum thrust with area ratio of 75:1 209,000-lb sea level thrust with area ratio of 35:1
Minimum Delivered Specific Impulse Efficiency	96% of theoretical shifting I_s at nominal thrust; 94% of theoretical shifting I_s during throttling
Throttling Range	Continuous from 100 to 20% of nominal thrust over the mixture ratio range
Overall Mixture Ratio Range	Engine operation from 5.0:1 to 7.0:1
Rated Chamber Pressure	2740 psia
Engine Weight (with 75:1 nozzle)	3520 lb (with flight-type actuators and engine command unit) 3380 lb (less flight-type actuators and engine command unit)
Expansion Ratio	Two-position booster-type nozzle with area ratios of 35:1 and 75:1
Durability	10 hours time between overhauls, 100 reuses, 300 starts, 300 thermal cycles, 10,000 valve cycles
Single Continuous Run Duration	Capability from 10 seconds to 600 seconds
Engine Starts	Multiple restart at sea level or altitude
Thrust Vector Control	Amplitude: ±7 deg; Rate: 30 deg/sec; Acceleration: 30 rad/sec^2
Control Capability	±3% accuracy in thrust and mixture ratio at nominal thrust. Excursions from extreme to extreme in thrust and mixture ratio within 5 seconds.

(C)(U) Table I. Demonstrator Engine Characteristics (Continued)

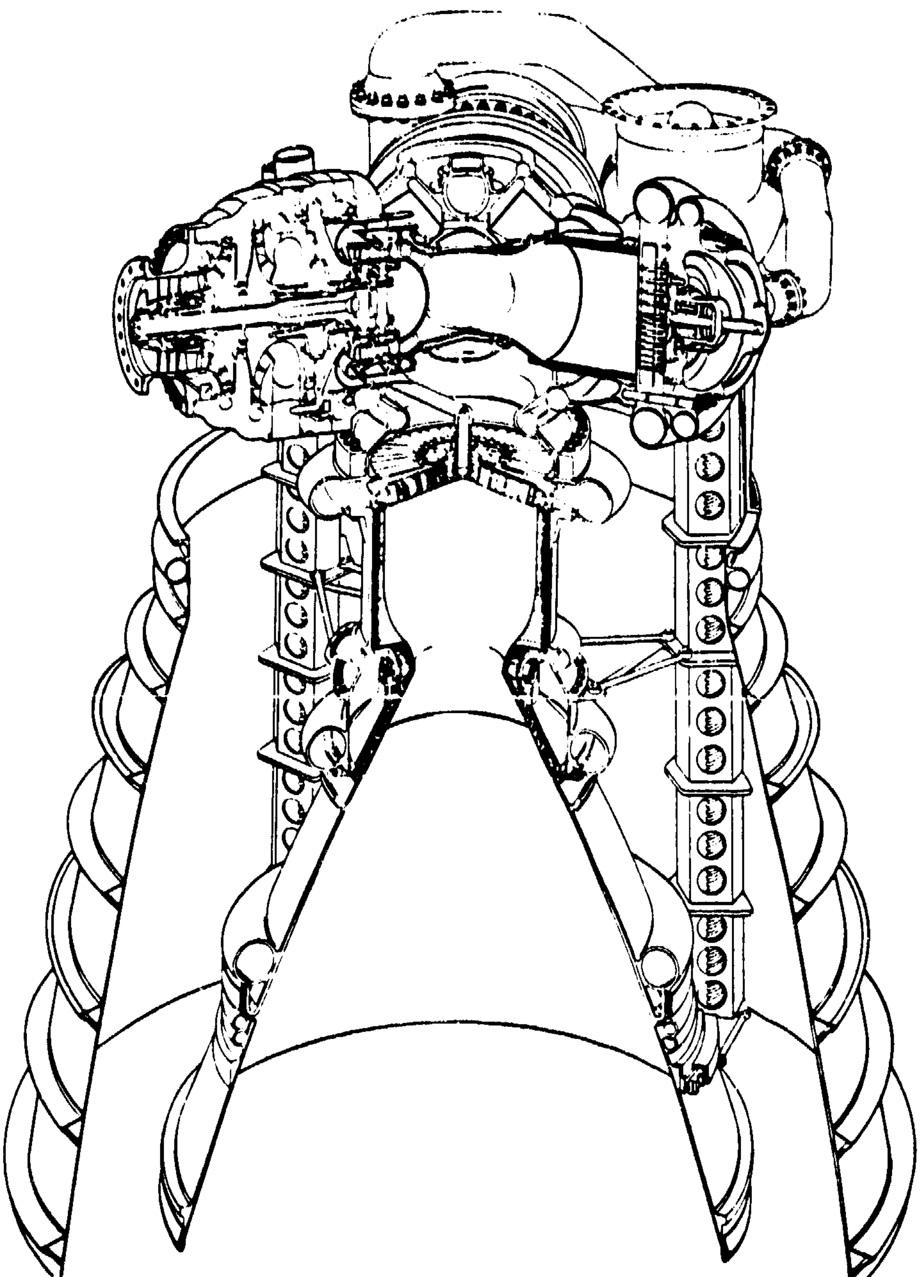
Propellant Conditions	LO_2: 16 ft NPSH from 1 atmosphere boiling temperature to 180°R LH_2: 60 ft NPSH from 1 atmosphere boiling temperature to 45°R
Environmental Conditions	Sea level to vacuum conditions Combined acceleration: 10 g's axial with 2 g's transverse, 6.5 g's axial with 3 g's transverse, 3 g's axial with 6 g's transverse
Engine/Vehicle	The engine will receive no external power, with the exception of normal electrical power and 1500-psia helium from the vehicle

B. ENGINE INSTALLATION DRAWING

(U) The general component arrangement of the XLR129-P-1 demonstrator engine is illustrated in Figure 1. An installation drawing with envelope dimensions, including the retracted (stowed) length and extended length of the engine, is provided as Figure 2. The primary interfaces, such as the propellant inlet connections and gimbal attachment locations, are also shown. It is estimated that the maximum actuator load during gimbaling of the demonstrator engine is 50,000 lb. The power to gimbal the engine is approximately 80 horsepower.

C. ENGINE FLOW SCHEMATIC

(U) A simplified propellant flow schematic illustrating the propellant flow paths and functional component arrangement of the engine is shown in Figure 3. The XLR129-P-1 high pressure rocket engine uses a staged combustion cycle in which most of the fuel is burned with a portion of the oxidizer in the preburner to provide turbopump power before combustion with the remainder of the oxidizer in the main burner chamber. Fuel and oxidizer enter the engine through the engine driven low-speed inducers. The low-speed inducers are used to minimize the fuel tank pressure requirements, while allowing high-speed main propellant pumps for high turbopump efficiencies. The fuel low-speed inducer is a single shaft unit with an axial flow inducer driven by a two-stage, axial-flow, partial-admission impulse turbine. The oxidizer low-speed inducer is also a single shaft unit with an axial flow inducer driven by a single stage radial inflow turbine.



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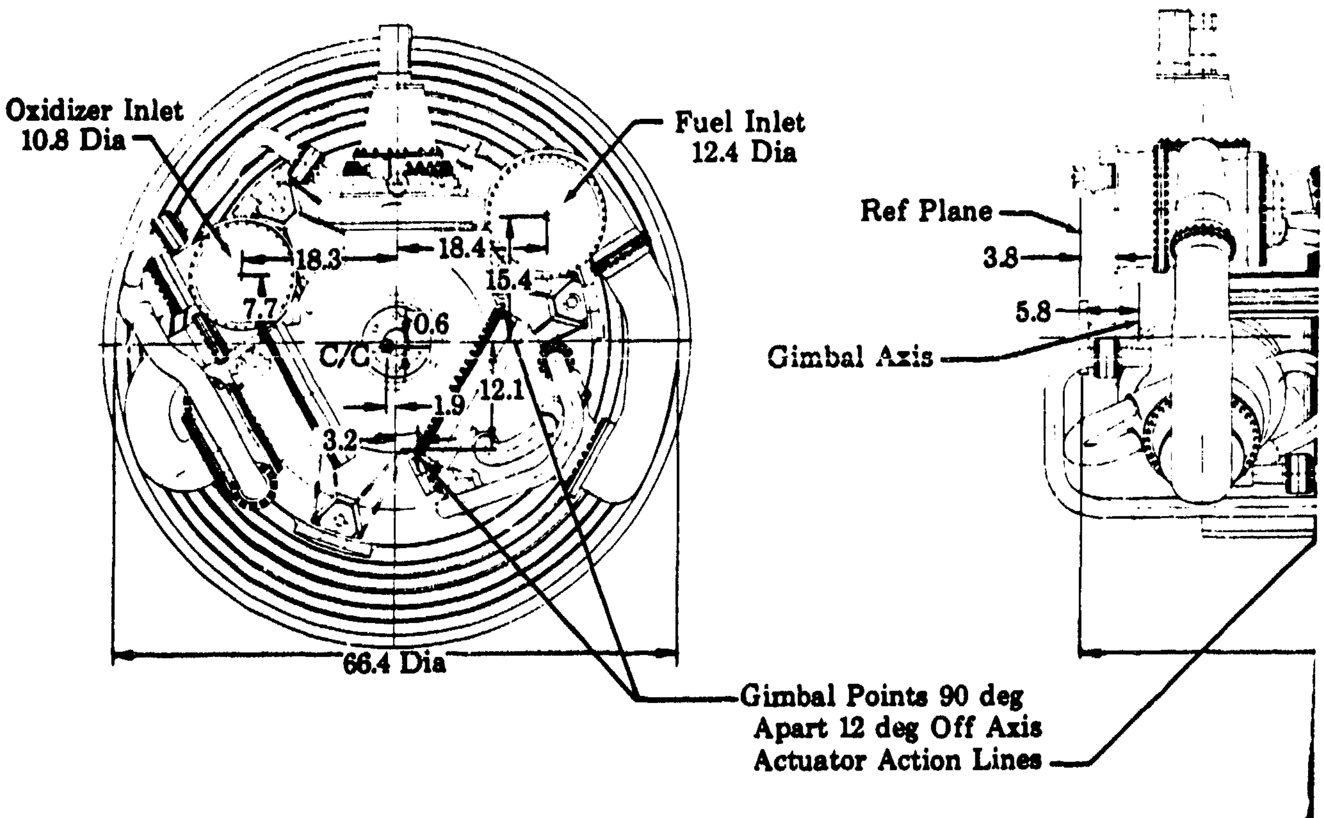
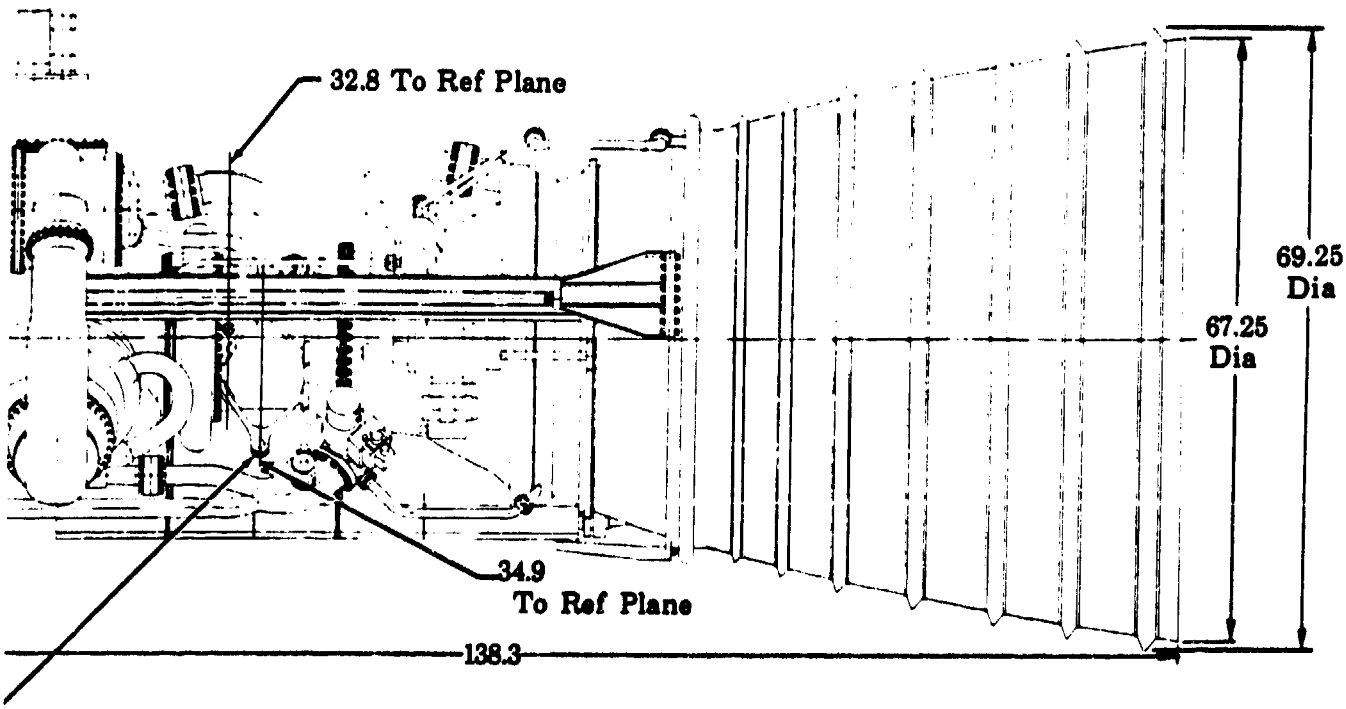
(U) Figure 1. XLR129-P-1 Demonstrator Engine FDC 27532

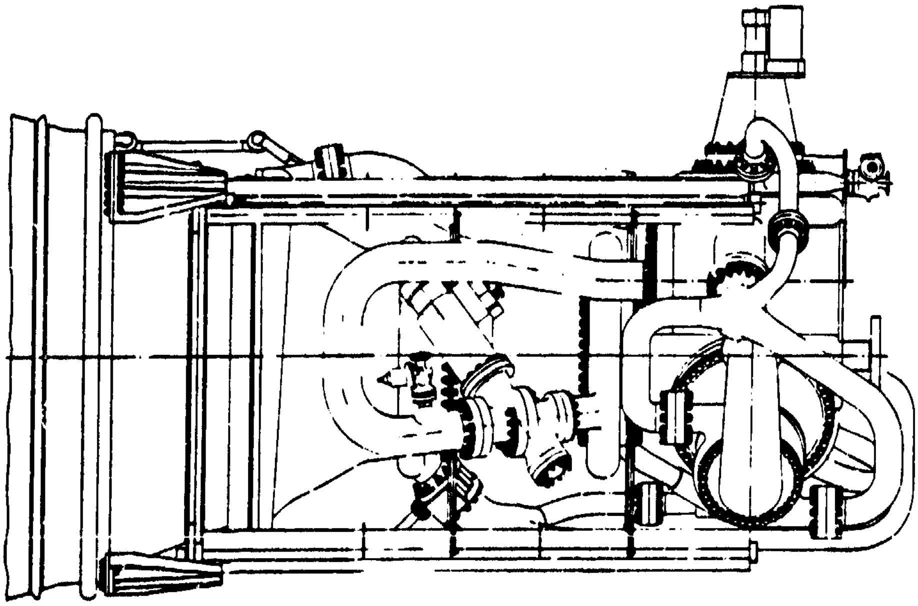


(U) Figure 2. Engine Installation Drawing

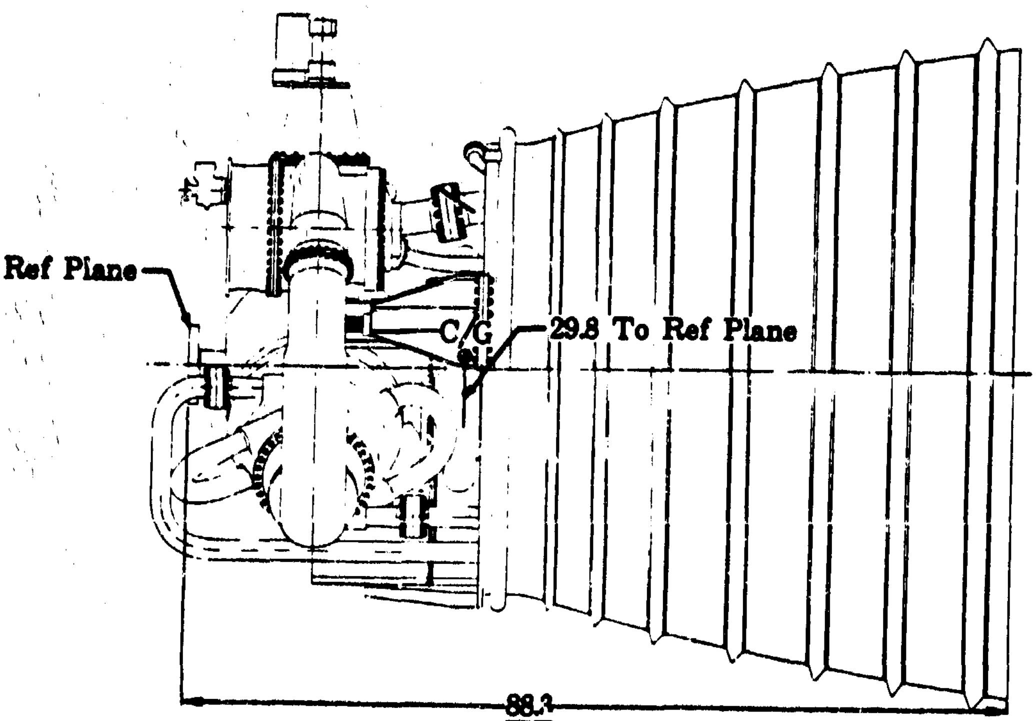


Nozzle Extended

FD 27724



(U) Figure 2. Engine Installation Drawing (Concluded)



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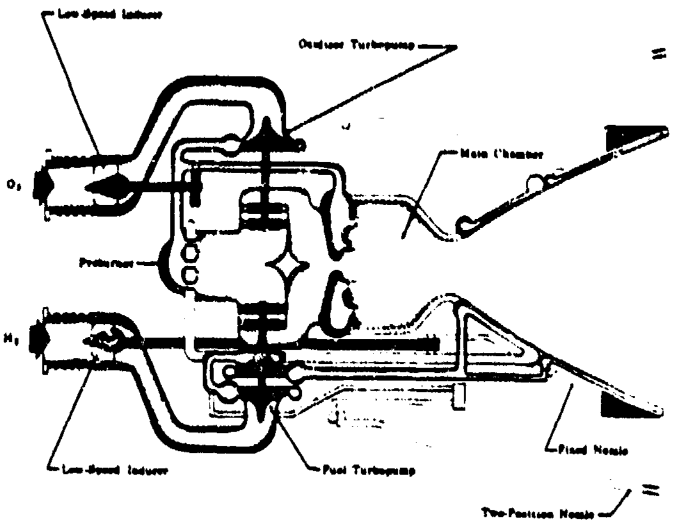


(U) Figure 3. Simplified XLR129-P-1 Demonstrator Engine Propellant Flow Schematic FD 21002C

(U) The main fuel turbopump is a single shaft unit with two back-to-back centrifugal pump stages driven by a two-stage, pressure-compounded turbine. The fuel flow is pumped to the system operating pressure levels by the main fuel pump. The hydrogen is then ducted to cool the regenerative sections of the nozzles. The principal fuel flow path from the pump is through the upstream portion of the primary nozzle, and then into the preburner chamber through the preburner injector. The remainder of the fuel flows through the downstream portion of the primary nozzle and then through the fuel low-speed inducer drive turbine prior to being passed into the main chamber as transpiration coolant. A small amount of fuel is bled off between the main fuel pump stages to provide coolant for the two-position nozzle. This coolant flows to the nozzle through a regulating orifice and a shutoff valve that is provided to stop the flow when the two-position nozzle is in the retracted position. The area ratio, at which the fixed primary nozzle ends and the two-position, translating nozzle starts, is varied to optimize the performance for the specific application.

(U) The oxidizer turbopump is a single shaft unit with a single, centrifugal pump stage driven by a two-stage, pressure-compounded turbine. After being pumped to the system operating pressure levels by the main oxidizer pump, the oxidizer is divided between the preburner and the main burner chamber. The principal oxidizer flow passes through and is the working fluid for the oxidizer low-speed inducer turbine before being injected into the main chamber. The remainder, a smaller scheduled portion of the oxidizer, is ducted to the preburner where it is burned with the fuel. The resulting combustion products flow through ducts to the two main turbines, which are arranged in parallel. The energy required to drive the main pumps is extracted from these combustion products, which then exhaust from the turbines and mix in a common passage of the transition case. These gases then pass through the main burner injector and into the main burner combustion chamber where they mix and burn with the principal oxidizer flow. These combustion gases are then expanded through the bell nozzle to provide thrust.

(U) The preburner injector consists of dual-orifice, tangential-swirler oxidizer injector elements and fixed area fuel injector elements. A preburner oxidizer valve is incorporated at the inlet to the injector assembly to vary the total oxidizer flow rate for turbine inlet temperature control and to adjust the relative flow of the primary and secondary oxidizer elements. The preburner combustion chamber is contained within the transition case, which also contains the turbine drive gas ducts. The main turbopumps are attached to the transition case with a single flange and bolt circle arrangement to provide ease of access for turbopump maintenance.

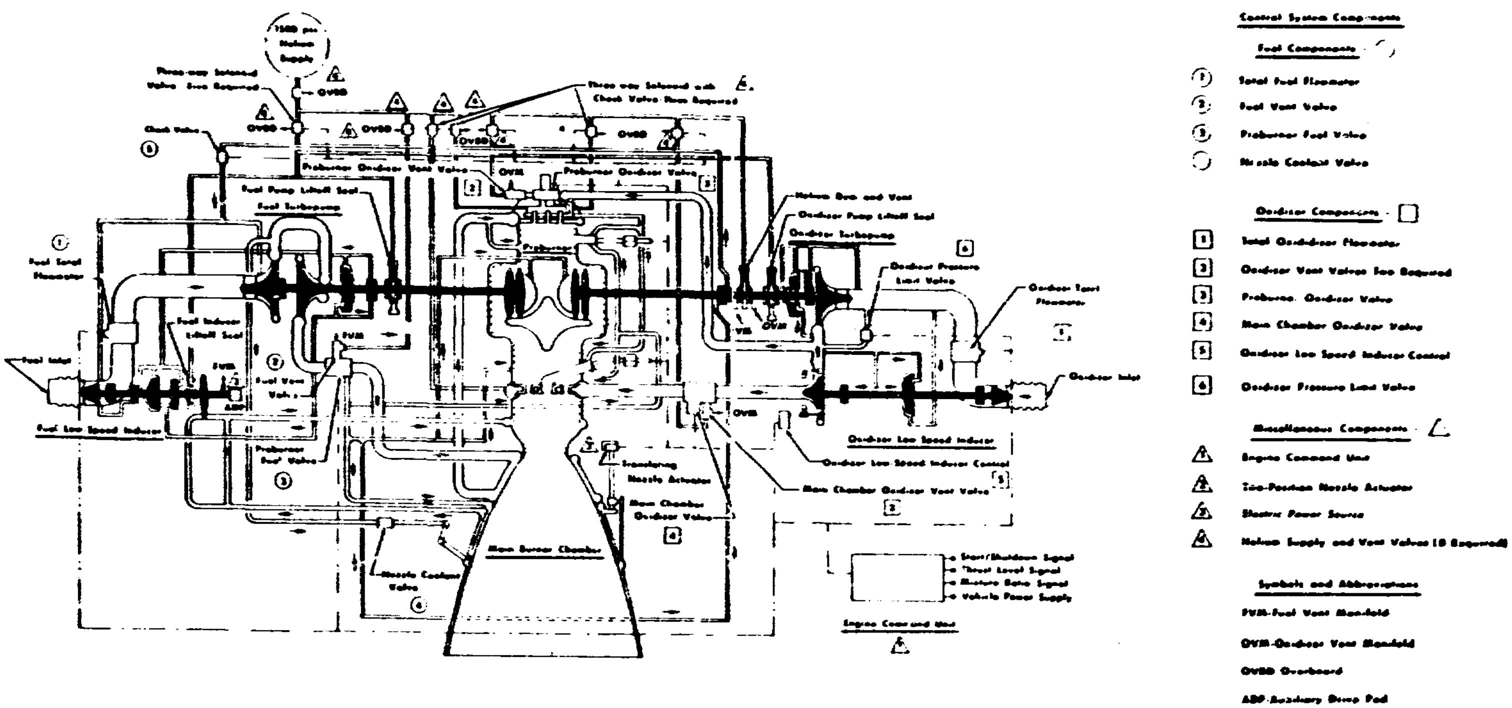
(C) The main chamber injector consists of fixed-area, tangential-swirler oxygen injection elements arranged in radial spraybars The fuel side (preburner gas after expansion through the turbine) is a fixed area design that ducts fuel-rich gas around each row of oxidizer injecter points. A small portion of the fuel-rich gas flows through a porous face to provide cooling. The combustion chamber wall is composed of a fuel cooled liner extending from the injector face through the throat region to a nozzle area ratio of 5.3. The liner is composed of porous wafer plates, which provides the transpiration cooling.

(U) The nozzle, which attaches downstream of the throat, is composed of two regeneratively cooled primary sections and a low-pressure, dump-cooled, two-position nozzle. The regeneratively cooled sections are conventional tubular heat exchangers, the two-position nozzle employs lightweight sheet metal construction.

(U) A more complete system schematic, including all main propellant lines, recirculation lines, electrical interconnections, and the helium systems is shown in Figure 4.

D. ENGINE AND COMPONENT OPERATING PARAMETERS

(C) The component and engine system steady-state operating parameters are presented in Table II for mixture ratios of 5, 6, and 7 at 100% and 20% thrust.



(U) Figure 4. Detailed XLR129-P-1 Demonstrator Engine Propellant Flow Schematic

GS 96903

(C)(U) Table II. XLR129-P-1 Demonstrator Engine Operating Characteristics, Booster

	100% Thrust r = 5.0	100% Thrust r = 6.0	100% Thrust r = 7.0	20% Thrust r = 5.0	20% Thrust r = 6.0	20% Thrust r = 7.0
Configuration						
Thrust, lb	244,000	244,000	244,000	48,800	48,800	48,800
Specific Impulse, sec	450	450	443	442	439	429
Envelope:						
Diameter, in.	69.25	69.25	69.25	69.25	69.25	69.25
Length: Nozzle Extended/Retracted, in.	138.3/88.3	138.3/88.3	138.3/88.3	138.3/88.3	138.3/88.3	138.3/88.3
Nozzle Area Ratio: Extended/Retracted	75/35	75/35	75/35	75/35	75/35	75/35
Fuel Flow, lb/sec	90.4	77.6	68.8	18.3	15.9	14.2
Oxidizer Flow, lb/sec	451.8	465.7	481.7	91.7	95.5	99.7
Total Propellant Flow, lb/sec	542.2	543.3	550.5	110.0	111.4	113.9
Main Burner Chamber						
Throat Total Pressure, psia	2805	2740	2676	552	537	526
Mixture Ratio (injector)	5.54	6.67	7.90	5.89	7.32	8.77
Specific Impulse Efficiency, %	96.8	97.0	95.9	96.4	96.1	96.0
Fuel Injector Pressure Loss, psi	161	130	120	16	16	17
Oxidizer Injector Pressure Loss, psi	822	913	984	40	42	46
Momentum Pressure Loss, psi	0.9	2.3	-1.7	9.4	7.8	7.3
Transpiration Coolant Flow, lb/sec	6.40	5.39	5.41	1.39	1.51	1.56
Throat Diameter, in.	7.68	7.68	7.68	7.68	7.68	7.68
Transpiration-Cooled Nozzle Section: Area Ratio	5.3	5.3	5.3	5.3	5.3	5.3
Preburner						
Total Pressure, psia	4824	4367	4175	732	719	710
Mixture Ratio (preburner injector)	1.07	1.1[illegible]	1.26	0.72	0.95	1.16
Temperature, °R	2015	2086	2326	1394	1796	2181
Fuel Injector Pressure Loss, psi	320	246	200	29	26	22
Oxidizer Injector and Control Valve Pressure Loss, psi	1101	981	566	533	596	635
Total Propellant Flow, lb/sec	155	136	126	23.0	21.4	20.4
Combustion Efficiency, %	100	100	100	100	100	100
Primary Nozzle						
Transpiration Supply Section (ε = 18 to 35):						
Coolant Flow, lb/sec	6.39	5.39	5.40	1.34	1.46	1.51
Coolant Inlet Pressure, psia	4903	4460	4516	1029	1116	1165
Coolant Inlet Temperature, °R	144	135	139	71	78	85
Coolant Pressure Loss, psi	76.9	62.1	63.7	11	12	13
Coolant Temperature Rise, °R	306	388	405	420	423	433
Preburner Supply Section (ε = 5.3 to 18):						
Coolant Flow, lb/sec	76	65	56	13.5	11.0	9.4
Coolant Inlet Pressure, psia	5293	4728	4468	770	752	738
Coolant Inlet Temperature, °R	141	133	140	71	77	82
Coolant Pressure Loss, psi	131	98	78	7	6	5
Coolant Temperature Rise, °R	42	51	57	55	70	84

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(C)(U) Table II. XLR129-P-1 Demonstrator Engine Operating Characteristics, Booster (Continued)

	100% Thrust r = 5.0	100% Thrust r = 6.0	100% Thrust r = 7.0	20% Thrust r = 5.0	20% Thrust r = 6.0	20% Thrust r = 7.0
Two-Position Nozzle (ϵ = 35 to 75)						
Coolant Flow, lb/sec	2.31	2.23	2.24	1.40	1.35	1.30
Thrust, lb	893	896	914	403	408	406
Fuel Turbopump						
Pump:						
Number of Pump Stages	2	2	2	2	2	2
Speed, rpm	48,043	44,477	44,220	19,650	20,947	21,933
Pressure Rise, psi	5542	4374	4911	991	1083	1136
Overall Efficiency, %	65.9	65.4	63.9	50.5	45.9	42.8
Impeller Tip Velocity (rms), ft/sec	2441	2261	2248	999	1065	1115
Temperature Rise, °R	91.1	82.5	87.1	24.4	31.7	38.2
Inlet Flow, lb/sec	91.3	79.5	69.7	18.7	16.3	14.6
Turbine:						
Number of Stages	2	2	2	2	2	2
Pressure Ratio	1.61	1.51	1.48	1.26	1.27	1.28
Inlet Temperature, °R	1986	2055	2292	1367	1760	2138
Inlet Pressure, psia	4766	4318	4129	726	712	703
Temperature Drop, °R	173	154	160	52.8	67.0	79.8
Mean Wheel Velocity, ft/sec	1488	1378	1370	609	649	679
Efficiency, %	75.4	75.2	75.0	63.5	62.7	62.3
Inlet Flow, lb/sec	111.5	97.3	90.0	16.5	15.4	14.6
Oxidizer Turbopump						
Pump:						
Number of Stages	1	1	1	1	1	1
Speed, rpm	25,263	23,263	22,369	10,099	10,290	10,400
Pressure Rise, psi	5737	5182	4603	1152	1204	1227
Efficiency, %	67.3	67.7	68.1	47.1	47.8	48.7
Impeller Tip Velocity, ft/sec	952	854	821	371	378	382
Temperature Rise, °R	35.0	31.1	27.4	10.4	10.7	10.7
Inlet Flow, lb/sec	630.6	535.7	548.1	127.0	131.4	135.9
Turbine:						
Number of Stages	2	2	2	2	2	2
Pressure Ratio	1.64	1.53	1.51	1.28	1.30	1.31
Inlet Flow, lb/sec	45.3	39.5	36.6	6.8	6.3	6.02
Inlet Temperature, °R	1986	2055	2292	1367	1760	2138
Inlet Pressure, psia	4782	4332	4141	727	714	705
Temperature Drop, °R	162.5	144	148	49	61	71
Mean Wheel Velocity, ft/sec	1131	1015	976	441	449	454
Efficiency, %	68.0	67.5	66.5	53.9	51.7	50.3

(C)(U) Table II. XLR129-P-1 Demonstrator Engine Operating Characteristics, Booster (Continued)

	100% Thrust r = 5.0	100% Thrust r = 6.0	100% Thrust r = 7.0	20% Thrust r = 5.0	20% Thrust r = 6.0	20% Thrust r = 7.0
Low-Speed Inducer						
Fuel Inducer:						
Flow Rate, lb/sec	90.4	77.6	68.8	18.3	15.9	14.2
Speed, rpm	19,777	17,846	16,908	6091	6094	6041
Pressure Rise, psi	90.2	94.7	108.9	25.0	29.0	31.0
NPSH, ft	60	60	60	60	60	60
Efficiency, %	80.0	79.2	77.3	48.5	42.9	38.7
Oxidizer Inducer:						
Flow Rate, lb/sec	451.8	465.7	481.7	91.7	95.5	99.7
Speed, rpm	5316	5221	5164	2008	2128	2236
Pressure Rise, psi	253	223	197	62	70	77
NPSH, ft	16	16	16	16	16	16
Efficiency, %	78.6	80.2	80.6	37.8	38.0	38.3
Fuel Low-Speed Inducer Turbine						
Pressure Ratio	1.38	1.34	1.36	1.37	1.40	1.41
Flow Rate, lb/sec	4.90	4.12	4.13	1.03	1.12	1.16
Speed, rpm	19,777	17,846	16,908	6091	6094	6041
Efficiency, %	60.5	55.0	56.9	28.3	27.1	26.0
Oxidizer Low-Speed Inducer Turbine						
Pressure Drop, psi	747	637	565	523	598	660
Flow Rate, lb/sec	371	393	411	81.4	84.3	87.9
Speed, rpm	5316	5221	5164	2008	2128	2236
Efficiency, %	51.9	51.1	50.2	34.1	33.8	33.8
Preburner Oxidizer Valve						
Inlet Pressure, psia	5885	5316	4707	1265.2	1314.8	1344.8
Inlet Temperature, °R	225.8	220.3	215.1	200.0	200.3	200.1
Exit Pressure, psia	5202	4564	4468	742.8	729.4	719.9
Flow, lb/sec	70.72	62.44	63.72	2.97	3.42	3.85
Overall Effective Area, $in.^2$	0.49	0.45	0.75	0.02	0.03	0.03
Main Chamber Oxidizer Valve						
Inlet Pressure, psia	5215	4735	4195	741.7	717.2	685.3
Inlet Temperature, °R	225.8	220.3	215.1	200.0	200.3	200.1
Exit Pressure, psia	3635	3663	3666	601.7	587.6	579.9
Flow, lb/sec	370.56	393.16	410.63	81.39	84.32	87.93
Overall Effective Area, $in.^2$	1.70	2.19	3.25	1.27	1.37	1.58

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(C)(U) Table II. XLR129-P-1 Demonstrator Engine Operating Characteristics, Booster (Continued)

	100% Thrust r = 5.0	100% Thrust r = 6.0	100% Thrust r = 7.0	20% Thrust r = 5.0	20% Thrust r = 6.0	20% Thrust r = 7.0
Preburner Fuel Valve						
Inlet Pressure, psia	5600	4956	5019	1061.5	1157.3	1212.28
Inlet Temperature, °R	138.7	130.1	135.12	70.84	78.5	85.32
Exit Pressure, psia	5300	4733	4472	770.2	751.7	738.0
Flow, lb/sec	81.29	69.69	60.84	15.41	12.88	11.21
Throat Effective Area, $in.^2$	3.44	3.44	1.93	0.70	0.52	0.43
Oxidizer Pressure Limit Valve						
Inlet Pressure, psia	6000	5412	4804	1266.8	1316.7	1346.9
Inlet Temperature, °R	225.8	220.3	215.1	290.0	200.3	200.1
Exit Pressure, psia	262	229.6	201	104.8	112.7	120.1
Flow, lb/sec	105.43	0.0	0.0	0.0	0.0	0.0
Overall Effective Area, $in.^2$	0.25	0.0	0.0	0.0	0.0	0.0

E. SUMMARY WEIGHT TABLE

(U) The estimated dry weights of the major components of the demonstrator and flight engines are shown in Table III. The demonstrator dry engine weights are based upon lightweight component designs with the additional margins required for a low risk demonstrator engine program. The flight engine dry weights are based upon the normal improvements and weight reduction that would materialize as a result of an engineering development program.

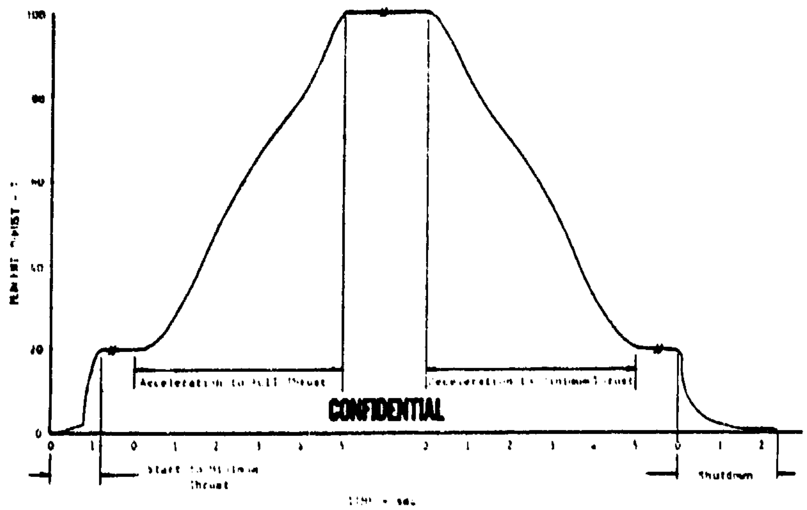
(C)(U) Table III. Engine Weights (Dry)

Item	Demonstrator Engine (ϵ = 75)	Flight Engine (ϵ = 75)
Preburner and Hardware	90	70
Transition Case and Gimbal	370	285
Main Burner Injector and Hardware	115	85
Main Burner Chamber	425	330
Nozzle and Actuation	640	460
Fuel Turbopump	480	380
Oxidizer Turbopump	335	250
Low-Speed Inducers	235	185
Controls	305	240
Plumbing	310	240
Miscellaneous	75	55
Total	3380*	2580

*Does not include valve actuators

F. START AND SHUTDOWN CURVE

(C) The start and shutdown transient curve from the XLR129-P-1 demonstrator engine is shown in Figure 5. This curve presents percent thrust versus time, and shows four distinct modes of operations (1) start to minimum thrust in approximately 1 second, (2) accelerations to 100% thrust, (3) decelerations to idle, and (4) shutdowns.



(U) Figure 5. Estimated Start and Shutdown Transient Characteristics DFC 69931

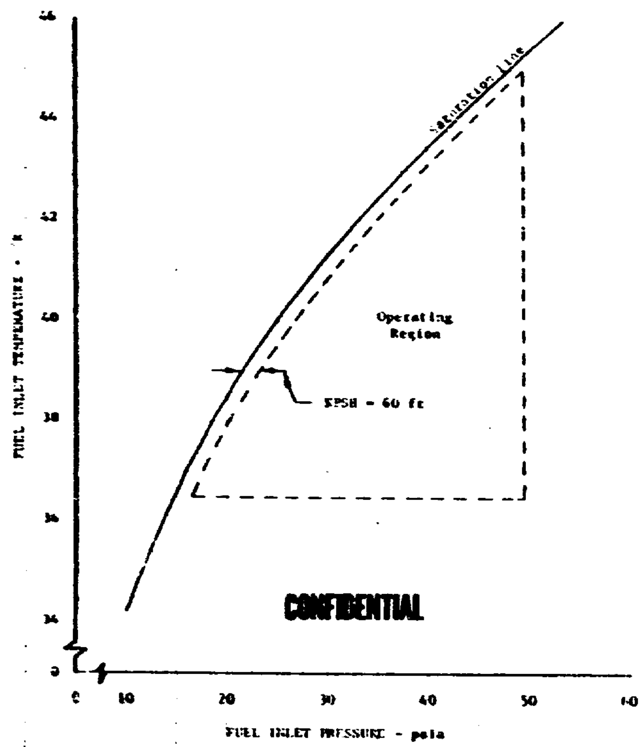
G. ENGINE INLET CONDITION CURVES

(U) The ranges of temperature, pressure and NPSH conditions required at the inlet to the fuel and oxidizer low-speed inducers are shown in Figures 6 and 7.

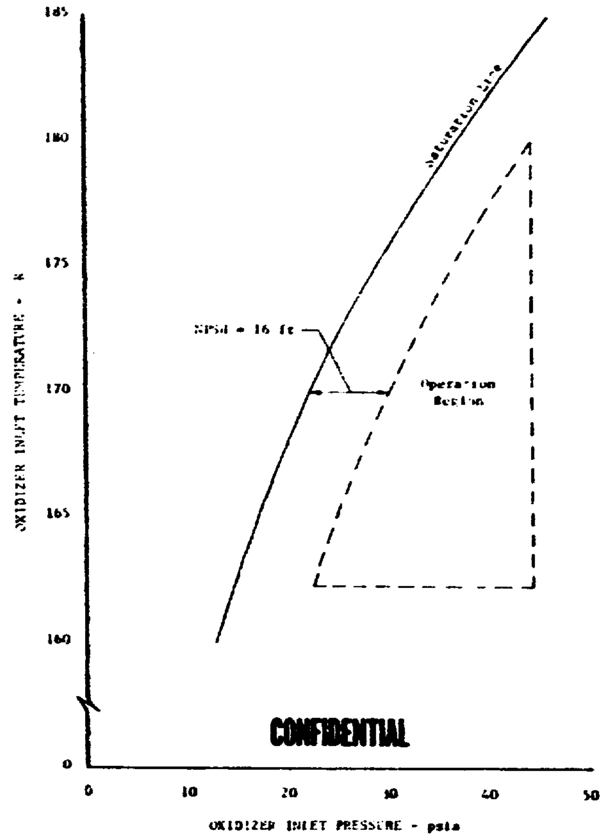
(U) The relationship required between fuel temperature and oxidizer temperature, so that the engine thrust and mixture ratio will remain at their set points within the specified control accuracy (±3%), is shown in Figure 8.

H. PROGRAM SCHEDULE

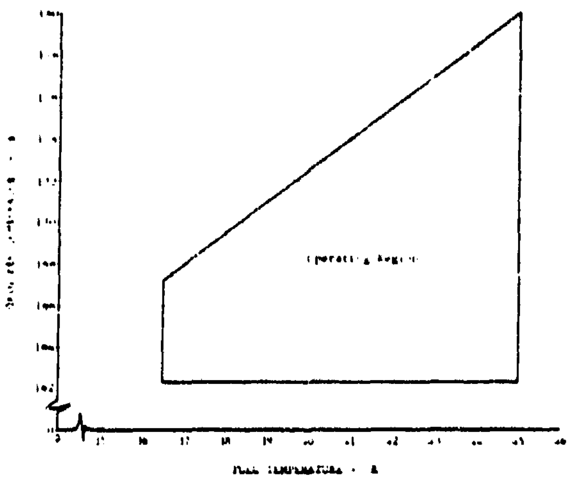
(U) The XLR129-P-1 demonstrator engine program schedule is shown in Figure 9. This is a 54-month program that began on 6 November 1967. The program has been divided into five phases. The first phase, which has already been completed, generated test and analytical data to complete the technology necessary to design the engine and components. During the second phase all the components will be designed. The components will be fabricated and tested to qualify them for engine use during the third phase. The fourth phase is the integration of the components into the demonstrator engine and the testing of the demonstrator engine. The fifth phase is the definition of the flight engine.



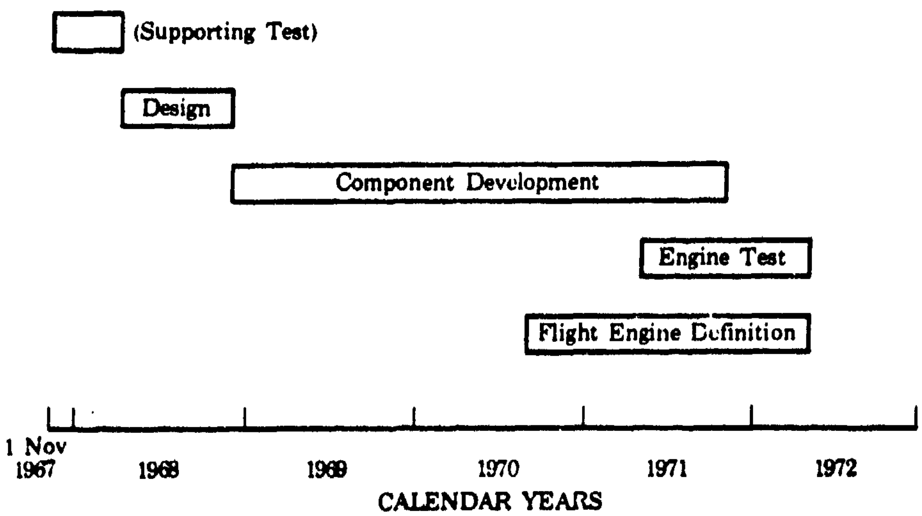
(U) Figure 6. Fuel Inlet Operating Region DFC 69932



(U) Figure 7. Oxidizer Inlet Operating Region DFC 69933



(U) Figure 8. Propellant Temperature Limits for Full Trim Capability DF 70473



(U) Figure 9. XLR129-P-1 Demonstrator Engine Program Schedule FD 27857

SECTION IV
FLIGHT ENGINE

SECTION IV
FLIGHT ENGINE

A. INTRODUCTION

(U) The flight engine configuration will be based on the results of the XLR129-P-1 demonstrator engine program. During a subsequent engineering development program further tests would be conducted to refine the engine to meet the flightweight criteria and to develop the level of system maturity required for flight operation. The additional development effort is a logical extension of the demonstrator program.

B. DESCRIPTION

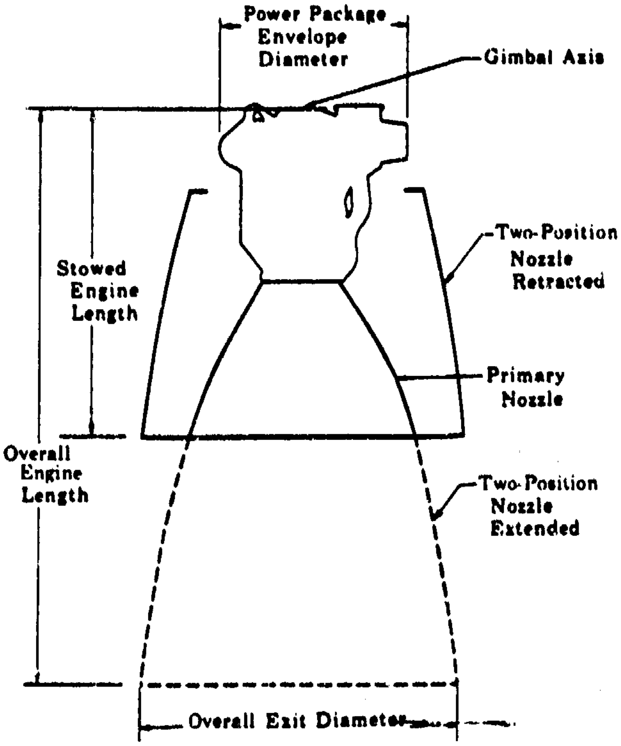
1. General

(C) The staged-combustion, high-pressure oxygen/hydrogen rocket engine is a versatile, high performance propulsion system for use in both upper and lower stages of advanced vehicles. This reusable engine is capable of maintaining constant thrust over a mixture ratio range of 5 to 7. Nozzle interchangeability and the use of the two-position nozzle concept permit operation of the same engine system with optimum nozzle area ratios for improving the performance of the lower stages within the atmosphere as well as providing the high performance attainable with very high area ratio nozzles in the upper stages. Interchangeability is achieved by attaching the desired nozzle to a fixed turbomachinery power package. The area ratio at which the primary (fixed) nozzle ends and the two position (translating) skirt starts, can be varied to optimize performance for the specific application. The general propellant flow schematic of the LR129-P-1 flight engine is the same as shown for the demonstrator engine in Figure 3. Engine nomenclature is illustrated in Figure 10.

2. Nozzle Concepts

a. Two-Position Nozzle Concept

(U) The two-position nozzle concept consists of a translating two-position nozzle and a fixed primary bell nozzle. The two-position nozzle is in the retracted position for sea level operation thus eliminating overexpansion losses associated with the larger area ratio section of the nozzle. This concept, combined with high chamber pressure, provides a compact engine package that provides superior low altitude performance. The two-position nozzle is translated to the extended position for high altitude operation. On upper stage applications, the two-position nozzle is in the retracted position before stage separation to provide a compact engine package, and is then extended after staging.



(U) Figure 10. Engine Configuration With Two-Position Nozzle FD 21152A

(C) Data for two types of two-position nozzles are presented in this report; (1) a configuration with a 35:1 primary nozzle expansion ratio and various overall expansion ratios and (2) alternatives which produce minimum stowed length of the engine with the nozzle retracted.

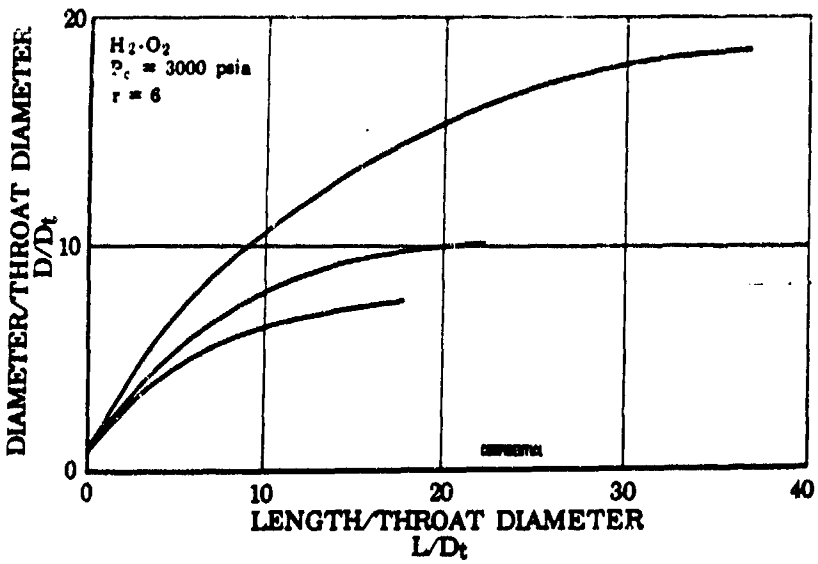
(C) In general, lower stage engines, which operate both at low-altitude and in vacuum, will have a relatively low primary expansion ratio. Pratt & Whitney Aircraft studies have shown the 35:1 primary expansion ratio to be near optimum. These studies have also shown that overall (extended) nozzle expansion ratios for these applications are also somewhat low and in the 50:1 to 150:1 range. High efficiency contours usually produce the best performance in these applications.

(C) When the primary nozzle expansion ratio ϵ_p (or the breakpoint between the stationary and the movable portions of the nozzle), is in the 35:1 to 80:1 range, it is possible to move the nozzle far enough forward to provide the altitude compensating feature. The two-position nozzle, however, cannot be moved completely forward to the gimbal axis because of interference with the turbopumps and plumbing. Where short length and high expansion ratios are primary installation goals (e.g., upper stage), it is generally desirable to accomplish the breakpoint at an area ratio greater than 80:1 so that the translating portion of the nozzle can be moved back as close as possible to the engine gimbal axis.

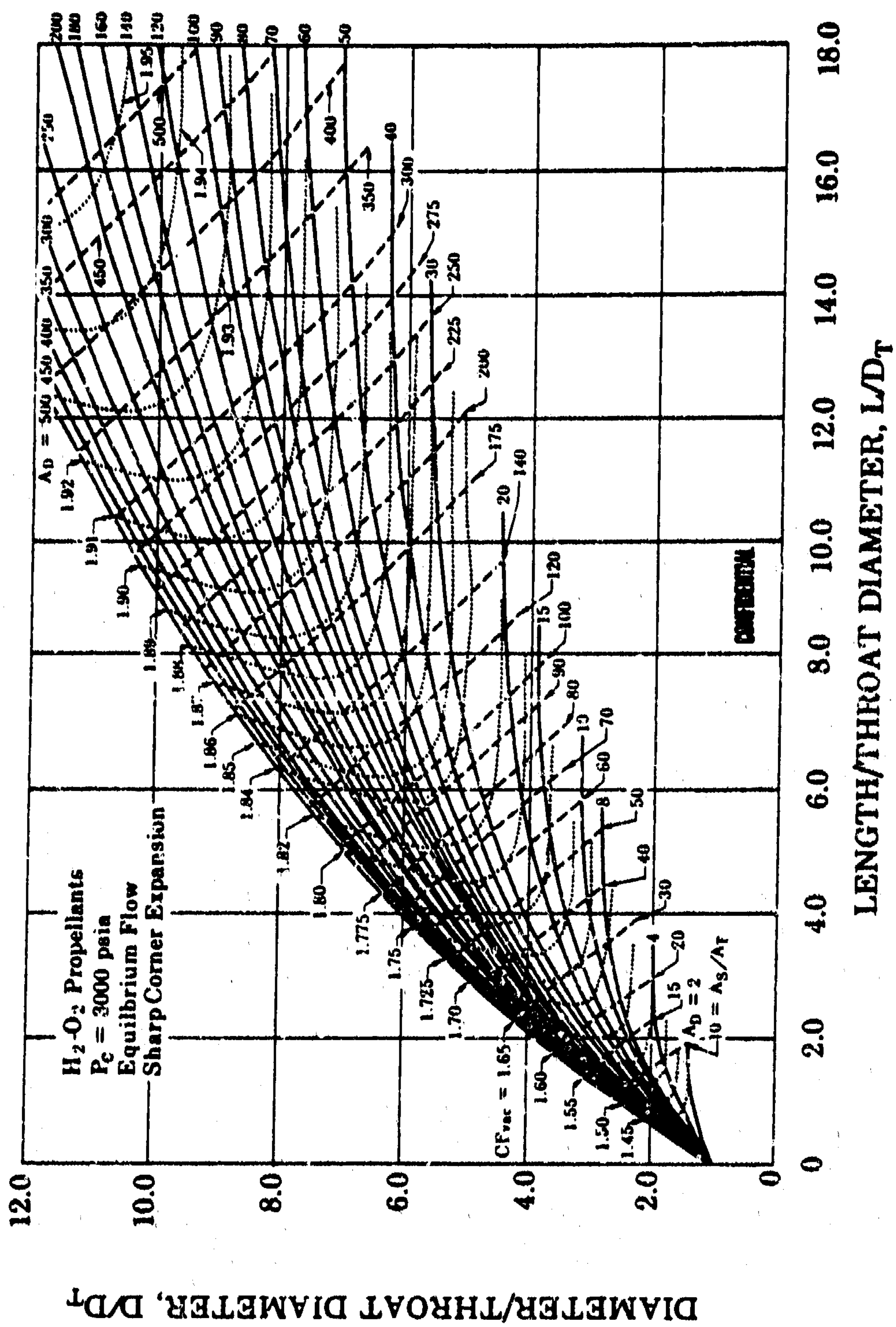
Because in upper stage applications the engine would not generally be fired with the nozzle retracted, the breakpoint could be selected to provide the shortest possible engine stowed length and the maximum engine expansion ratio. The engine stowed length curves presented in this report reflect these considerations.

b. Nozzle Contour

(U) Engine length, weight, and performance are functions of the exhaust nozzle contour. In general, shorter length contours yield lower nozzle performance. Therefore, in addition to optimization of the area ratio, the optimization of a bell nozzle engine includes the selection of the shape or contour of the nozzle. The bell nozzle contours used by Pratt & Whitney Aircraft are selected from a family of truncated perfect nozzles. Perfect nozzles are defined as those that, at a prescribed area ratio, expand a gas flow from the throat of the nozzle to a uniform and parallel flow at the nozzle exit. Using the method of characteristics, a series of perfect nozzle contours may be computed as a function of this "design" area ratio. The contour surface at any diameter and length along the nozzle may be plotted in nondimensional form as shown in Figure 11. The integrated thrust and surface area can also be calculated at axial locations along the nozzle. The calculation procedure includes the effect of friction and varying thermodynamic properties of the reacting gases. Representative results of this detailed analysis can be plotted as shown in Figure 12, which presents contour coordinates for perfect bell nozzles with lines of constant surface area and constant vacuum thrust coefficients superimposed.



(U) Figure 11. Perfect Nozzle Contours FDC 27862



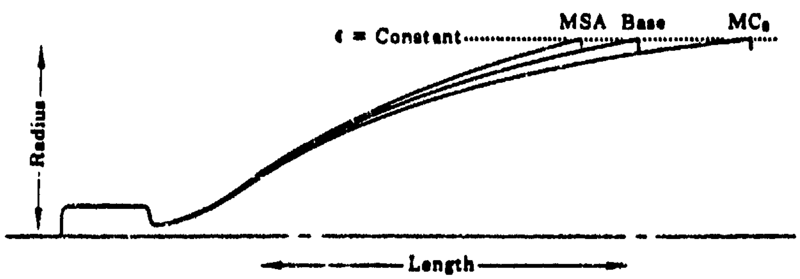
FD 23119

(U) Figure 12. Perfect Nozzle Contours

(U) A perfect nozzle contour does not necessarily produce optimum engine or vehicle performance. Because a perfect nozzle is constrained to produce completely axial flow at the exit, a considerable part of the rear section of the nozzle is involved in the final flow turning process. In a real nozzle, the friction losses here are greater than the performance gains that accrue from the final flow straightening. Therefore, maximum nozzle performance is obtained by shortening or truncating the perfect nozzle. Further, in real vehicles there are engine length penalties such that additional truncation may be required to produce maximum vehicle performance. Pratt & Whitney Aircraft has defined a series of three nozzle length truncations of the above defined perfect nozzle for application to real vehicles. These truncations (referred to as nozzle contours in this report) are described as follows:

1. Maximum Performance Nozzle (MC_s)
2. Minimum Surface Area Nozzle (MSA)
3. Base Nozzle.

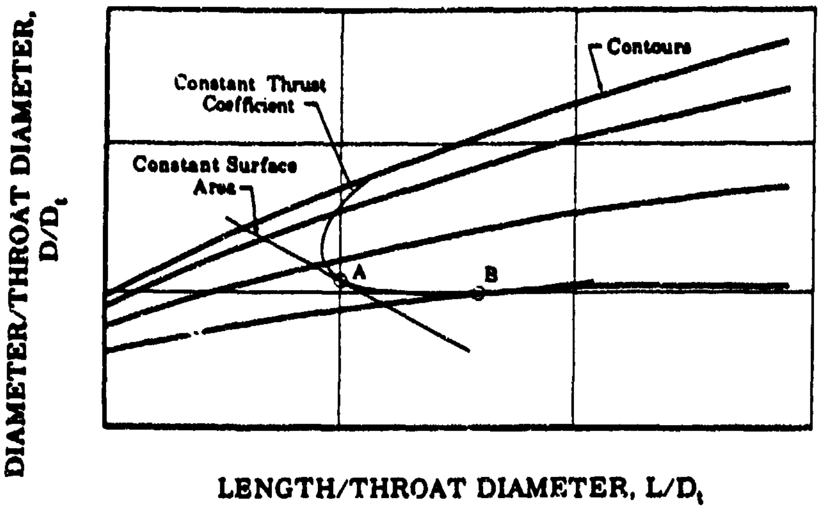
These three nozzle truncations are shown schematically in Figure 13.



(U) Figure 13. Nozzle Contour FD 22438A

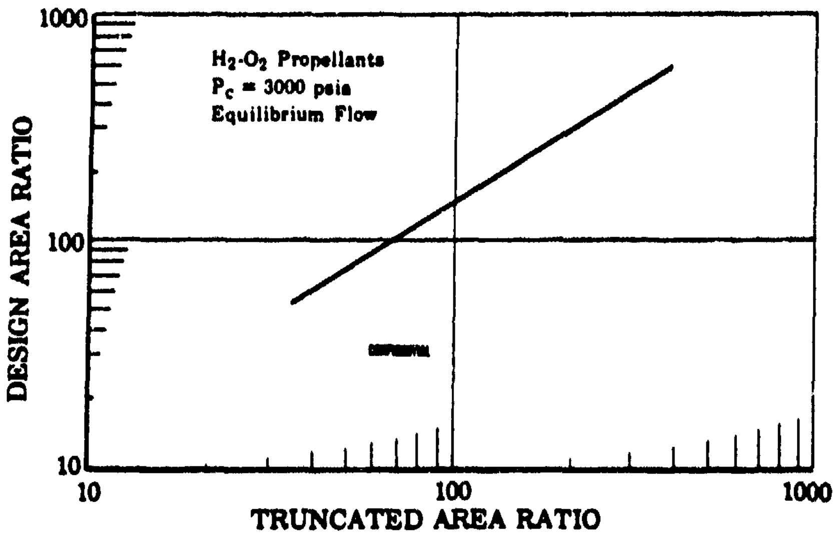
(U) So that the method used in establishing MC_s, MSA, and Base Nozzle contours can be more easily identified, a representative portion of the information given in Figure 12 is shown in Figure 14.

(U) Nozzles with minimum surface area (MSA nozzles) for a given thrust are defined by the locus of points for which a line of constant thrust coefficient ($C_{F_{vac}}$) and a line of constant surface area-to-throat area ratio (A_s/A_t) are tangent. This is shown as point A on Figure 14. Maximum nozzle efficiency (MC_s) for a given thrust are defined by the locus of points for which the lines of constant thrust coefficient ($C_{F_{vac}}$) have zero slope (point B on Figure 14). It should be noted that the maximum performance nozzle (point B) is still short of the full length perfect nozzle because frictional drag has been included.



(U) Figure 14. Contour Optimization FD 6263C

(U) A third type of nozzle truncation considered is referred to as a base nozzle contour. This truncation has resulted from experience with various optimization studies and generally produces nearly optimum balance of weight and performance, particularly for lower stage applications. While this contour is not established directly from analysis of Figure 14, it falls approximately half-way between points A (MSA) and B (MC_s). The relationship between truncated area ratio and "design" area ratio for base nozzle contours is shown in Figure 15.



(U) Figure 15. Base Truncations of Perfect Nozzles FDC 27861

(U) Nozzle truncations considered in the parametric data of this report are: maximum performance, base, and minimum surface area nozzle contours. Pratt & Whitney Aircraft preliminary engine/vehicle studies have shown that the best upper stage performance is generally obtained by using the base nozzle or minimum surface area nozzle contour.

3. Flight Engine Characteristics

(U) Preliminary design characteristics for the flight engine are provided in Table IV.

(C)(U) Table IV. Flight Engine Characteristics

Nominal Thrust	To be determined from parametric data
Minimum Delivered Specific Impulse Efficiency	100% nominal thrust - 96.7% 20% nominal thrust - 95.4% (All of these conditions are with an expansion ratio of 100:1, mixture ratio of 7 and MSA nozzle)
Throttling Range	Continuous from 100 to 20% of nominal thrust over the mixture ratio range
Overall Mixture Ratio Range	Engine operation from 5.0:1 to 7.0:1
Rated Chamber Pressure	3000 psia
Durability	10 hours time between overhauls, 100 reuses, 300 starts, 300 thermal cycles, 10,000 valve cycles
Single Continuous Run Duration	Capability from 10 seconds to 600 seconds
Engine Starts	Multiple restart at sea level or altitude
Thrust Vector Control	Amplitude: ±7 deg; Rate: 30 deg/sec; Acceleration: 30 rad/sec^2
Control Capability	±3% accuracy in thrust and mixture ratio at nominal thrust. Excursions from extreme to extreme in thrust and mixture ratio within 5 seconds
Propellant Conditions	LO_2: 16 ft NPSH from 1 atmosphere boiling temperature to 180°R LH_2: 60 ft NPSH from 1 atmosphere boiling temperature to 45°R

(C)(U) Table IV. Flight Engine Characteristics (Continued)

Environmental Conditions	Sea level to vacuum conditions Combined acceleration: 10 g's axial with 2 g's transverse, 6.5 g's axial with 3 g's transverse, 3 g's axial with 6 g's transverse
Engine/Vehicle	The engine will receive no external power, with the exception of normal electrical power and 1500-psia helium from the vehicle

4. Cycle Balance

(C) A 250K LR129-P-1 flight engine system and component operating parameters at 100% and 20% thrust and mixture ratios of 5, 6, and 7 are presented in Table V.

5. Schedule With Cost

(U) The schedules of the flight engine development program to PFRT as related to the XLR129-P-1 demonstrator engine program are shown in Figure 16. The estimated development costs to PFRT and Qualification as a function of rated thrust between 100K and 500K are shown in Figure 17.

C. PARAMETRIC ENGINE DATA

1. General

(U) Parametric engine data are presented in this section for two basic configurations: (1) engines equipped with two-position exhaust nozzles and (2) engines using conventional fixed exhaust nozzles.

(C) The two-position nozzle engines may be configured to provide either (1) high sea level performance or (2) minimum stowed length; the selection of the primary nozzle expansion ratio (ϵ_p) depending upon the application. The above objectives are achieved by having the two-position nozzle retract from a nozzle expansion ratio of 35 (denoted by $\epsilon_p = 35$) mainly for lower stage applications or from the expansion ratio that will provide minimum stowed length (denoted by ϵ_p = minimum).

(C) Engine data provided for the fixed and two-position nozzle engine configurations are: (1) performance (specific impulse), (2) weight, and (3) envelope. These data are for a vacuum thrust range of 100,000 lb (100K) to 500,000 lb (500K), mixture ratios from 5 to 7, and nozzle expansion ratios (ϵ) 35 to 400 for three nozzle truncations or contours), as applicable. Nozzle truncations considered in these data presentations are maximum performance (MC_s), base, and minimum surface area (MSA) nozzle contours. A discussion of nozzle contours is provided in Section II. Engine dimensions and nomenclature are illustrated in Figure 10.

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(C)(U) Table V. Estimated Flight Engine Operating Characteristics Upper Stage: Nozzle Extended

	100% Thrust r = 5.0	100% Thrust r = 6.0	100% Thrust r = 7.0	20% Thrust r = 5.0	20% Thrust r = 6.0	20% Thrust r = 7.0
Configuration						
Thrust, lb	250,000	250,000	250,000	50,000	50,000	50,[illegible]
Specific Impulse, sec	465	463	456	464	461	453
Mixture Ratio (overall)	5.0	6.0	7.0	5.0	6.0	7.0
Envelope:						
Diameter, in.	100	100	100	100	100	100
Length: Nozzle Extended/Retracted, in.	166/90	166/90	166/90	166/90	166/90	166/90
Nozzle Area Ratio: Extended/Retracted	184/80	184/80	184/80	184/80	184/80	184/80
Fuel Flow, lb/sec	89.6	77.1	68.5	18.0	15.5	13.8
Oxidizer Flow, lb/sec	448.0	462.8	479.7	89.8	93.0	96.6
Total Propellant Flow, lb/sec	537.6	539.9	548.2	107.8	108.5	110.4
Main Combustion Chamber						
Throat Total Pressure, psia	3091	3000	2925	615	593	577
Mixture Ratio (injector)	5.28	6.40	7.51	5.32	6.47	7.60
Transpiration Coolant Flow, lb/sec	2.3	2.5	2.6	0.6	0.6	0.7
Throat Diameter, in.	7.3	7.3	7.3	7.3	7.3	7.3
Preburner Combustion Chamber						
Total Pressure, psia	5014	4730	4493	803	771	742
Mixture Ratio (preburner injector)	1.02	1.18	1.34	0.65	0.80	0.94
Temperature, °R	1841	2086	2327	1215	1474	1693
Total Propellant Flow, lb/sec	156	141	129	23.3	21.0	19.2
Thrust Nozzle						
Regenerative Section:						
Exit Diameter, in.	43	43	43	43	43	43
Dump Section:						
Exit Diameter, in.	99.0	99.0	99.0	99.0	99.0	99.0
Coolant Flow, lb/sec	2.6	2.5	2.2	0.6	0.5	0.5
Fuel Turbopump						
Pump:						
Number of Pump Stages	2	2	2	2	2	2
Speed, rpm	48,000	47,400	46,500	21,000	21,400	21,100
Pressure Rise, psi	5591	5804	5713	1143	1170	1133
Overall Efficiency, %	71.4	70.7	69.6	57.8	53.3	50.4
Impeller Tip Velocity (rms), ft/sec	2300	2273	2230	1007	1027	1012
Flow Rate, lb/sec	87.0	74.7	66.3	17.4	15.0	13.2
Turbine:						
Number of Stages	2	2	2	2	2	2
Pressure Ratio	1.54	1.50	1.46	1.26	1.24	1.22
Inlet Temperature, °R	1841	2086	2327	1215	1474	1693
Efficiency, %	74.0	74.0	73.9	67.1	66.4	65.9
Flow Rate, lb/sec	103	93	85	15.5	13.9	12.7

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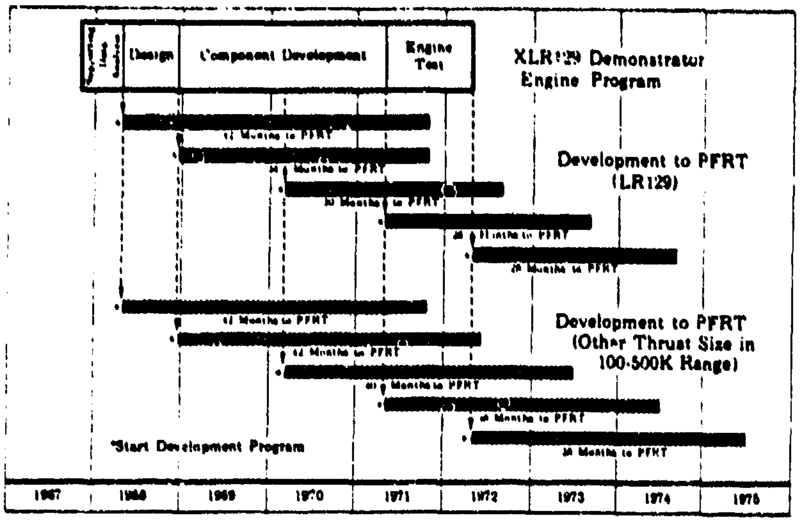
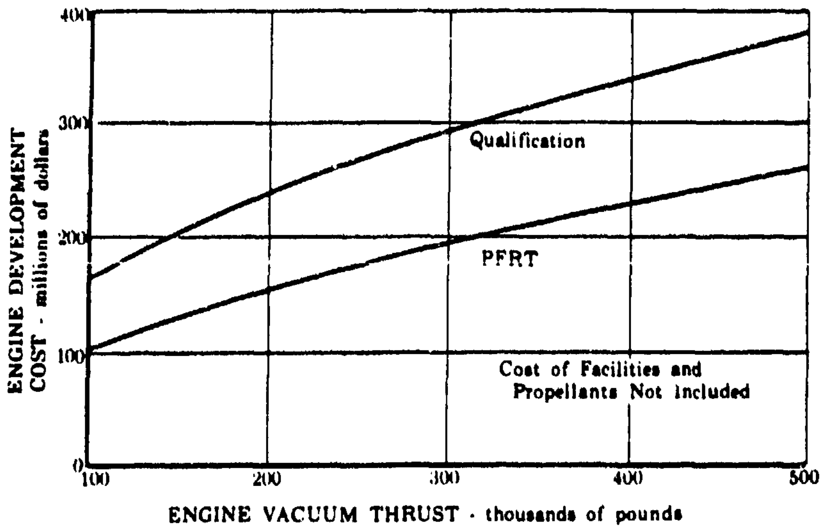
(C)(U) Table V. Estimated Flight Engine Operating Characteristics Upper Stage: Nozzle Extended (Continued)

	100% Thrust r = 5.0	100% Thrust r = 6.0	100% Thrust r = 7.0	20% Thrust r = 5.0	20% Thrust r = 6.0	20% Thrust r = 7.0
Oxidizer Turbopump						
Pump:						
Number of Stages	1	1	1	1	1	1
Speed, rpm	27,100	25,500	24,000	11,200	11,200	10,500
Pressure Rise, psi	6695	5902	5080	1248	1195	1096
Efficiency, %	72.2	72.8	73.0	50.8	52.6	55.0
Impeller Tip Velocity, ft/sec	923	867	816	381	374	358
Flow Rate, lb/sec	448.0	462.8	479.7	89.8	93.0	96.6
Turbine:						
Number of Stages	2	2	2	2	2	2
Pressure Ratio	1.42	1.39	1.36	1.19	1.19	1.18
Flow Rate, lb/sec	53	48	44	7.9	7.1	6.5
Inlet Temperature, °R	1841	2086	2327	1215	1476	1693
Efficiency, %	71.9	71.4	70.5	61.8	59.9	58.3
Low-Speed Inducer						
Fuel Inducer:						
Flow Rate, lb/sec	89.6	77.1	68.5	18.0	15.5	13.8
Speed, rpm	20,500	18,900	17,700	6773	6620	6240
Pressure Rise, psi	52.5	61.3	64	17.6	18.8	17.3
NPSH, ft	60	60	60	60	60	60
Efficiency, %	79.3	78.3	76.7	46.4	40.8	37.7
Oxidizer Inducer:						
Flow Rate, lb/sec	448.0	462.8	479.7	89.8	93.0	96.6
Speed, rpm	5503	5421	5380	1883	1826	1721
Pressure Rise, psi	178	162	150	31	28	26
NPSH, ft	16	16	16	16	16	16
Efficiency, %	78.8	80.3	80.6	39.4	41.5	44.6
Fuel Low-Speed Inducer Turbine						
Pressure Ratio	1.28	1.31	1.32	1.40	1.43	1.42
Flow Rate, lb/sec	4.1	4.5	4.5	1.1	1.2	1.2
Speed, rpm	20,500	18,900	17,700	6770	6620	6260
Efficiency, %	49.6	47.6	46.6	19.2	19.2	19.0
Oxidizer Low-Speed Inducer Turbine						
Pressure Drop, psi	588	541	507	206	170	126
Flow Rate, lb/sec	369	386	405	80.4	83.4	87.0
Speed, rpm	5500	5421	5380	1883	1826	1721
Efficiency, %	45.8	44.5	43.4	41.7	43.7	46.6



(U) Figure 16. XLR129-P-1 Demonstrator Engine Program FD 24282A



(U) Figure 17. Estimated Engine Development Costs, Oxygen/Hydrogen Engines FD 27781

(C) The engine characteristics of specific impulse, weight, and envelope are based on the following:

- Performance attainable at the time of preliminary flight rating test (PFRT)
- Engine inlet propellant conditions:

  Minimum required hydrogen net positive section head (NPSH) = 60 ft

Minimum required oxygen net positive suction head (NPSH) = 16 ft

- Continuous throttling capability between 100% and 20% of rated thrust
- Mixture ratio range of 5 to 7 at all thrust levels
- Thrust vector control provided by mechanical gimbaling, ±7 degrees
- Durability of 10 hours time between overhaul (TBO), 100 reuses, 300 starts, 300 thermal cycles, and 10,000 valve cycles
- Lightweight, dump-cooled nozzle construction
- Performance based on the use of nozzle dump cooling for expansion ratios greater than 35
- For high expansion ratio nozzles, radiation cooling is used aft of the lowest expansion ratio permitted by heat flux levels. (This expansion ratio varies over the parametric range, but is approximately 200.) If radiation cooled nozzle skirts were not used, an insignificant increase in engine weight would result.

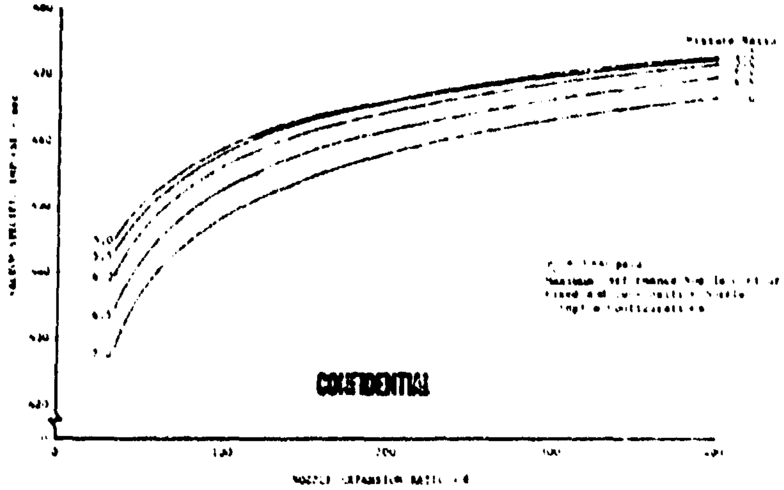
2. Engine Performance

a. General

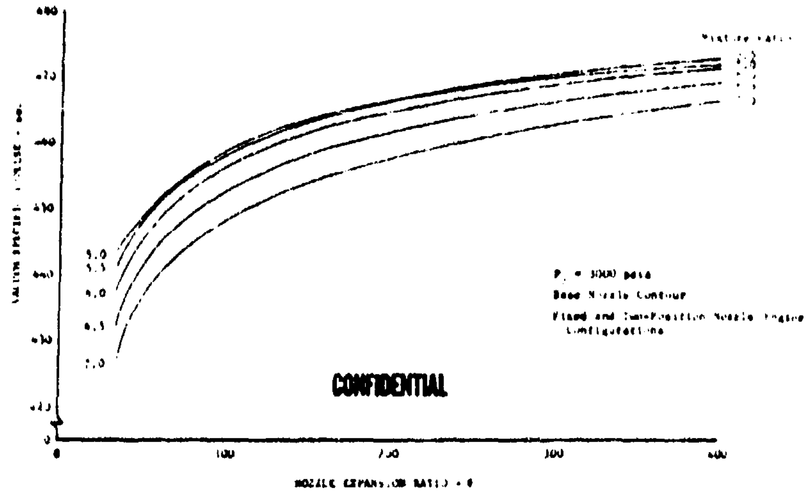
(U) Vacuum specific impulse, sea level specific impulse, altitude performance, and throttling performance are presented in this section for fixed and two-position nozzle engine configurations.

b. Vacuum Performance

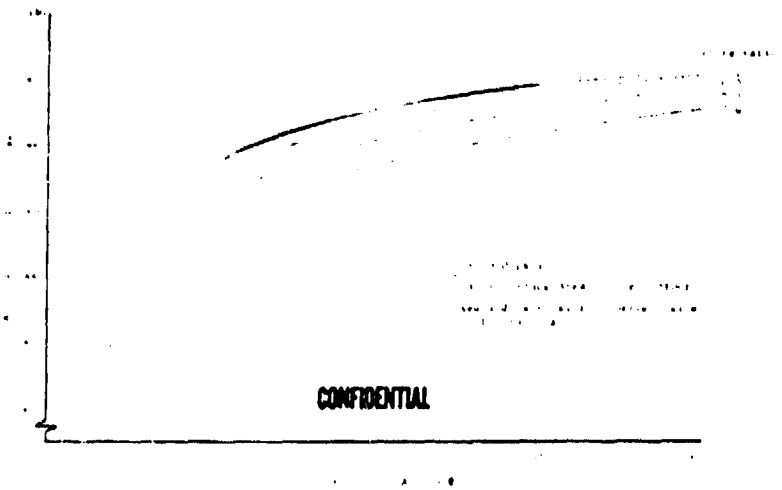
(C) Vacuum specific impulse data are presented in Figures 18 through 20 as a function of nozzle expansion ratio. Data are presented for maximum performance (MC_s), base, and minimum surface area (MSA) nozzle contours. These curves, which cover a nozzle expansion ratio range of 35 to 400, are applicable for all lightweight-nozzle (fixed and two-position) engine configurations in which dump cooling begins at an expansion ratio of 35:1. Vacuum specific impulse is very nearly independent of thrust level in the range of 100K to 500K; this is particularly so between 200K and 500K.



(U) Figure 18. Vacuum Specific Impulse vs Nozzle Expansion Ratio DFC 70293



(U) Figure 19. Vacuum Specific Impulse vs Nozzle Expansion Ratio DFC 70296



(U) Figure 20. Vacuum Specific Impulse vs Nozzle Expansion Ratio   DFC 70299

c. Altitude Performance

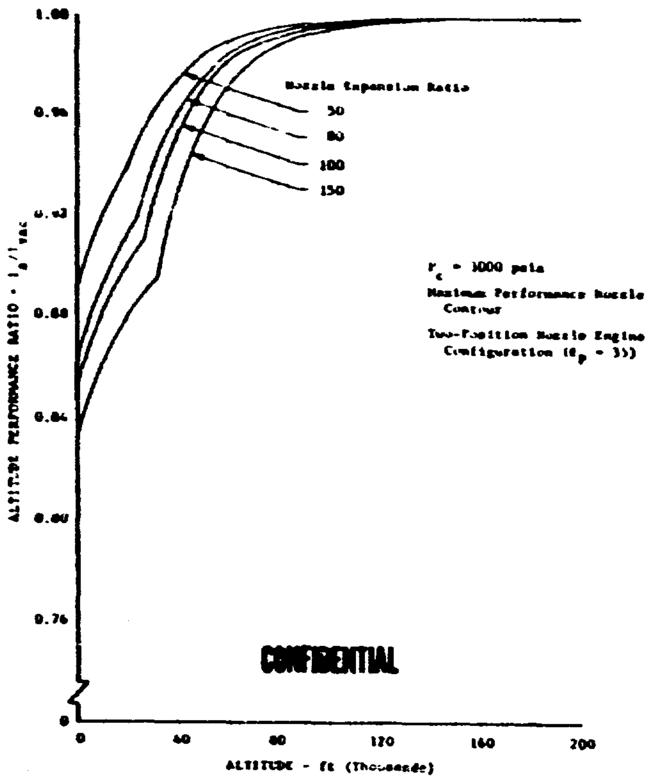
(1) General

(U) Altitude performance for engines equipped with lightweight nozzles is shown in Figures 21 through 26. These data are presented as the ratio of specific impulse (at altitude) to vacuum specific impulse ($I_{s_{alt}}/I_{vac}$), versus altitude as a function of nozzle expansion ratio. The altitude range is from sea level to 200,000 ft (vacuum conditions). The thrust at any altitude may also be obtained by use of these curves because $I_{s_{alt}}/I_{vac} = F_{alt}/F_{vac}$. Estimated nozzle flow separation altitude is also shown, where applicable, for the higher expansion ratios.

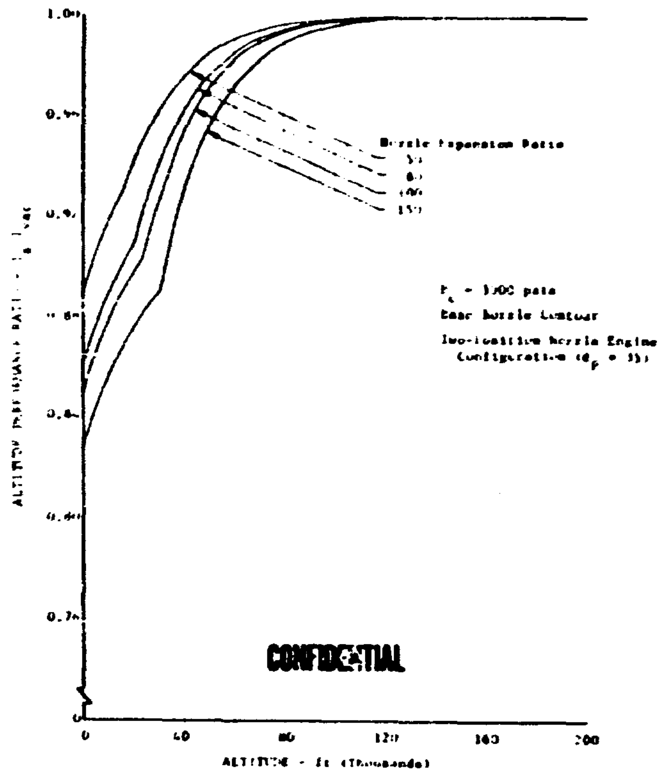
(2) Two-Position Nozzle Engine Configurations

(a) Primary Nozzle Area Ratio = 35

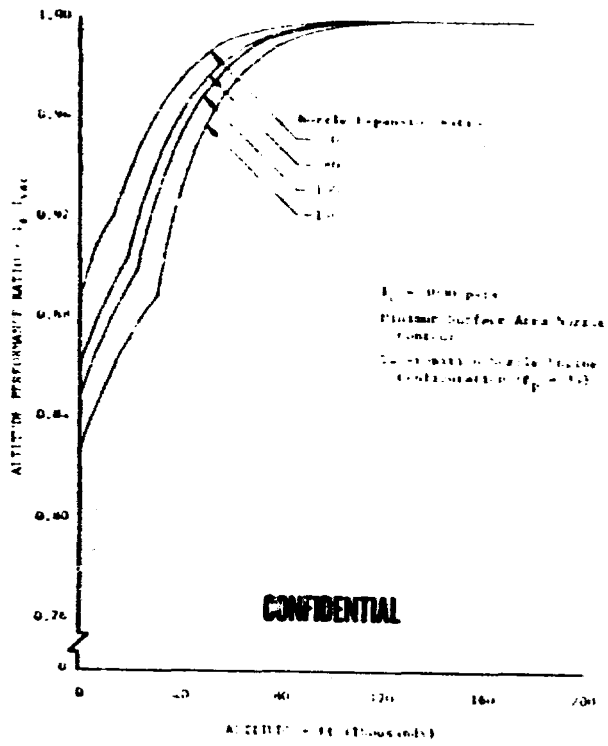
(C) Altitude performance for two-position nozzle engine configurations having a primary nozzle expansion ratio of 35 ($\epsilon_p = 35$) is shown in Figures 21 through 23 for MC_s, base, and MSA nozzle contours. These curves present altitude performance for the engines with the two-position nozzle in the extended position for high altitude operation and with it retracted for operation at lower altitudes. The inflection points in these figures result from the translation of the two-position nozzle. These data are usable for the entire parametric range of mixture ratios without significant error. These data are independent of thrust level for a constant primary expansion ratio. The nozzle expansion ratio range provided in these curves is from $\epsilon = 50$ to $\epsilon = 150$.



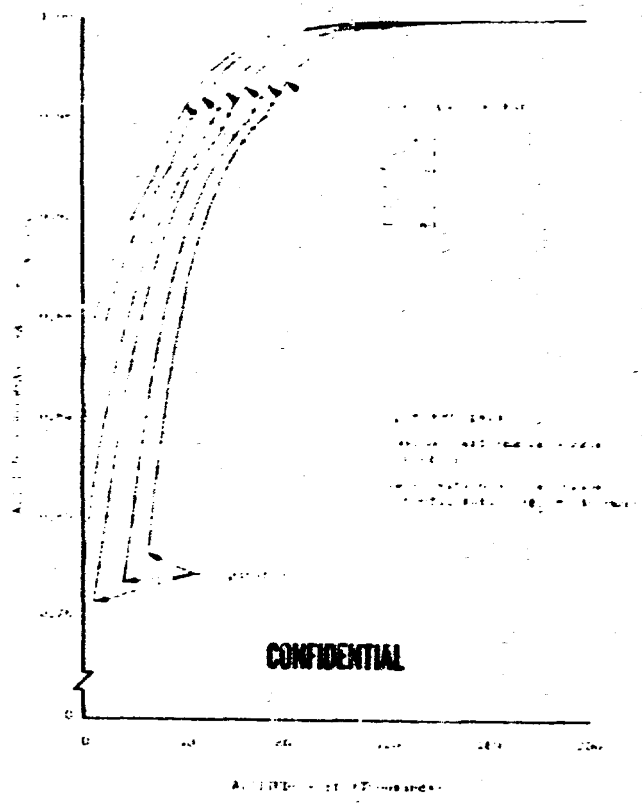
(U) Figure 21. Altitude Performance DFC 70271



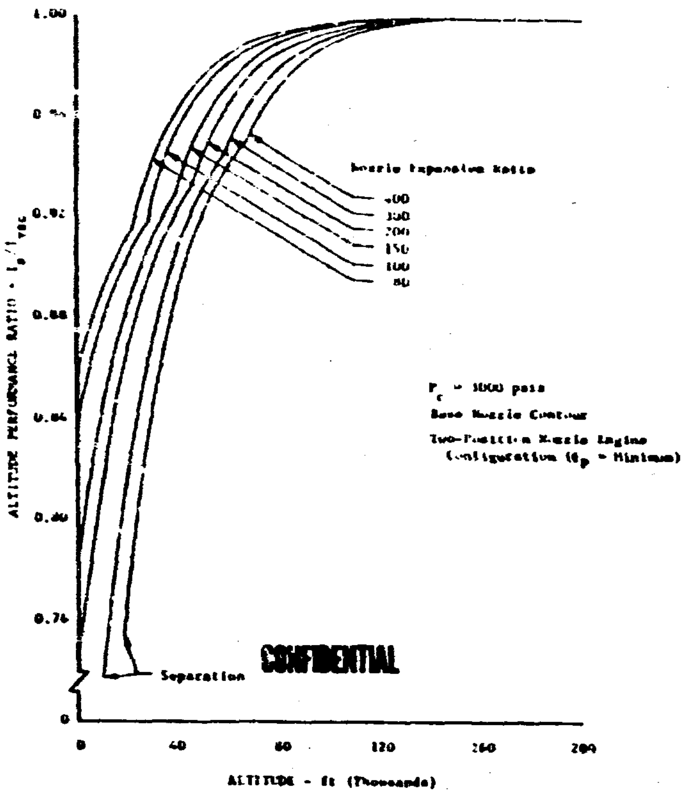
(U) Figure 22. Altitude Performance DFC 70270



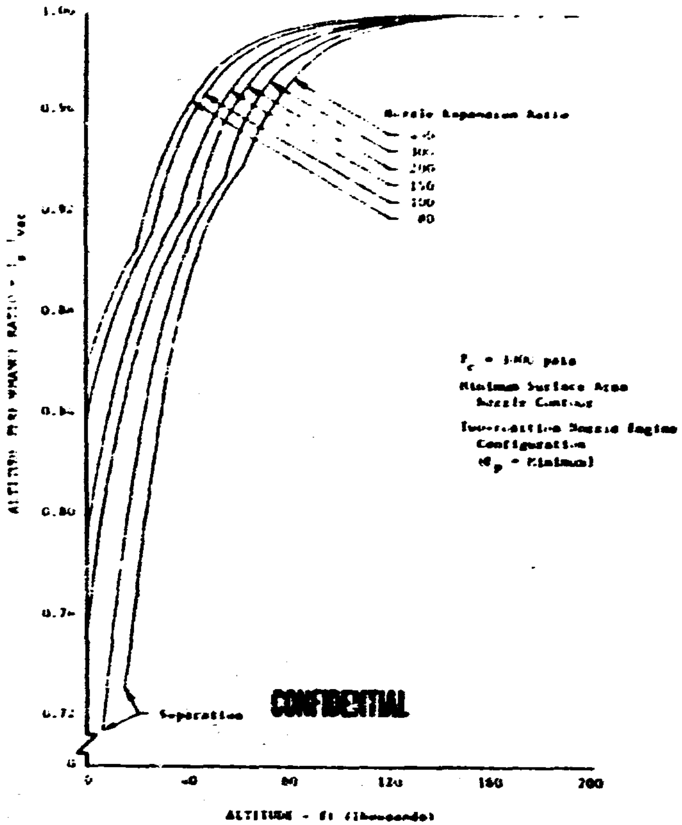
(U) Figure 23. Altitude Performance DFC 69853



(U) Figure 24. Altitude Performance DFC 69936



(U) Figure 25. Altitude Performance DFC 69934



(U) Figure 26. Altitude Performance DFC 69935

(b) Primary Nozzle Area Ratio Sized for Minimum Stowed Length

(U) Altitude performance for two-position nozzle engine configurations that provide minimum stowed length (ϵ_p = minimum) is presented in Figures 24 through 26.

(C) As in the case of two-position nozzles with a constant primary nozzle expansion ratio (ϵ_p = 35), the variation in altitude performance resulting from mixture ratio is insignificant. There is an additional effect caused by the variation in the primary nozzle expansion ratio as a function of thrust level. This effect is also generally small and produces a total error, including mixture ratio effects, which is no more than ±0.5% in sea level performance. Variations in primary nozzle expansion ratio also affect the secondary nozzle translation altitude (the altitude at which the secondary nozzle is translated to its extended position for high performance operation); the resultant effect in translation altitude is approximately ±3000 feet. Because all the variations cited above are relatively small, a single curve for r = 6.0 is presented for altitude performance for each nozzle contour. Nozzle expansion ratios for these curves cover the range from ϵ = 50 to ϵ = 400.

(3) Fixed Nozzle Engine Configurations

(C) Altitude performance for fixed nozzle engine configurations is presented in Figure 27 for nozzle expansion ratios of ϵ = 35 to ϵ = 400. It can be seen from this figure that lower stage applications with nozzle expansion ratios greater than approximately 100:1 are not practical with fixed nozzles because of nozzle flow separation at low altitudes.

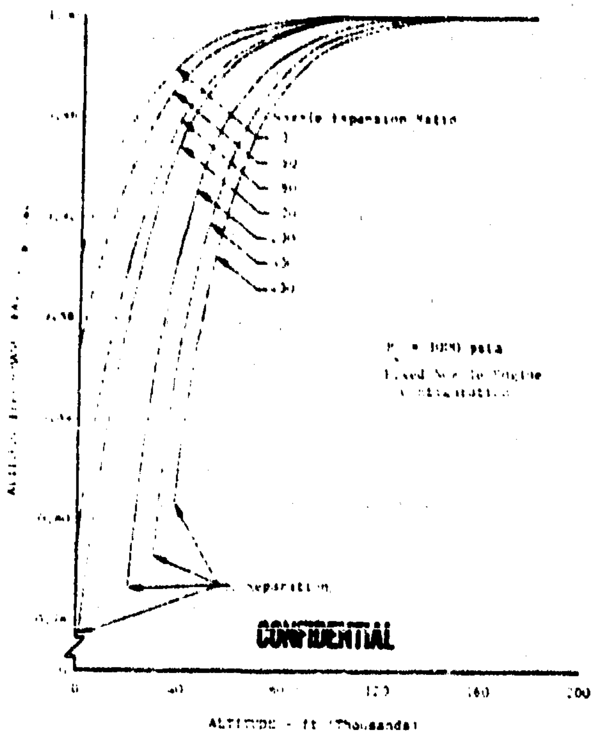
J. Sea Level Performance

(1) General

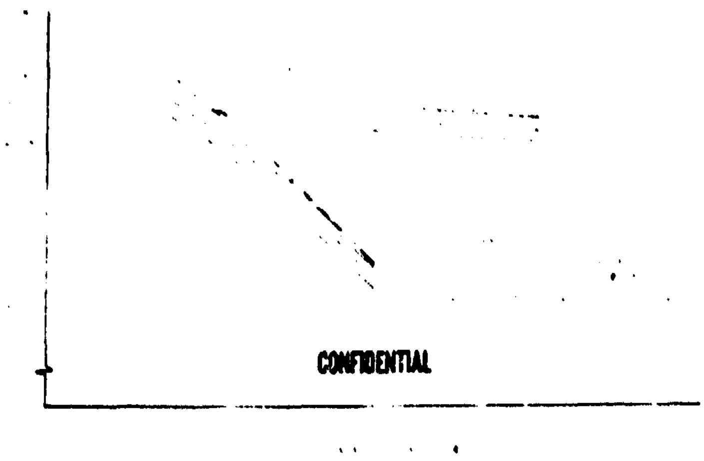
(C) To facilitate vehicle sizing and to allow a direct comparison of sea level performance, sea level specific impulse is presented in Figures 28 through 30 as a function of overall nozzle expansion ratio. Data are provided for three nozzle contours for fixed nozzle and two-position nozzle (ϵ_p = 35) engine configurations.

(2) Two-Position (ϵ_p = 35) and Fixed Nozzle Engine Configurations

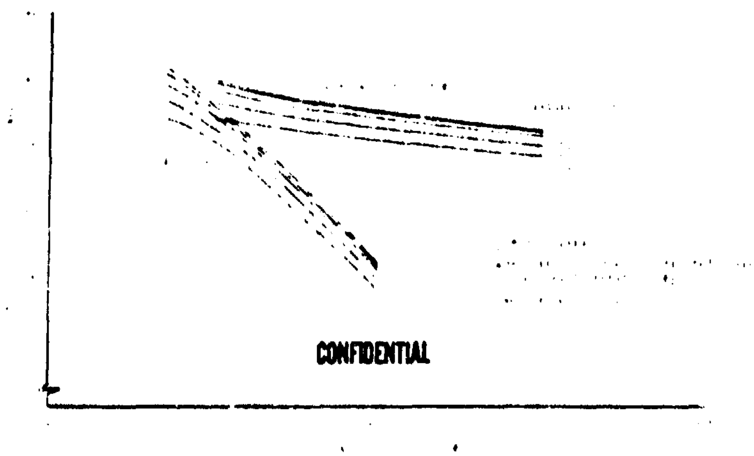
(C) Sea level specific impulse for two-position nozzle engine configurations having a primary nozzle expansion ratio of 35 is presented for overall expansion ratios of 50 to 150. Sea level specific impulse for fixed nozzle engine configurations is presented for expansion ratios 35 to 100. Flow separation at sea level occurs in fixed nozzles with expansion ratios approximately 100 or greater.



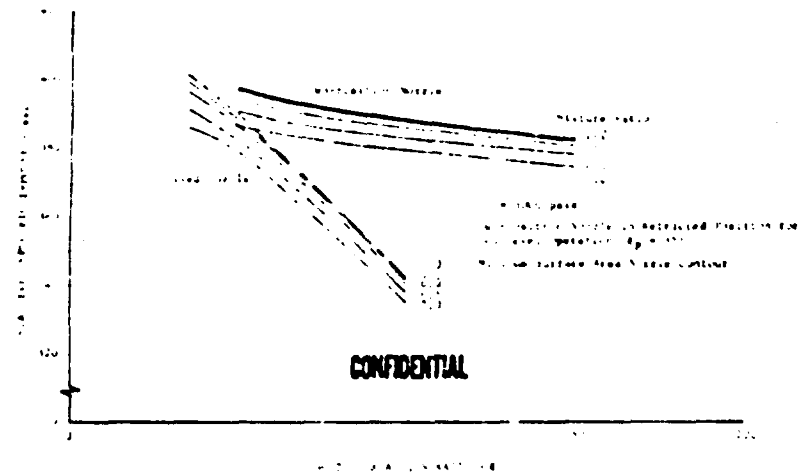
(U) Figure 27. Altitude Performance DFC 69930



(U) Figure 28. Sea Level Specific Impulse vs Nozzle Expansion Ratio DFC 70352



(U) Figure 29. Sea Level Specific Impulse vs Nozzle Expansion Ratio DFC 70354

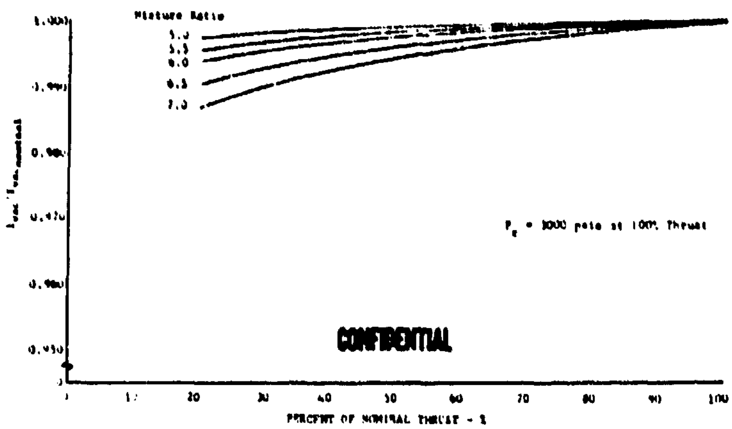


(U) Figure 30. Sea Level Specific Impulse vs Nozzle Expansion Ratio DFC 70357

(C) A comparison of the sea level specific impulse attainable with fixed nozzle and two-position nozzle engine configurations shows the significant improvement in performance that results from the use of a two-position nozzle. In the case of two-position nozzle engine configurations for use in lower stage applications, the translating nozzle is in the retracted position for sea level and low altitude operation. With the two-position nozzle retracted, the engine operates with a low expansion ratio (ϵ_p = 35). This produces a higher sea level specific impulse than that obtainable with a fixed nozzle having the same overall nozzle expansion ratio. A two-position nozzle engine configuration with an overall nozzle expansion ratio of 100 and a primary nozzle expansion ratio of 35 will provide a sea level specific impulse that is approximately 50 seconds greater than that attainable with a fixed nozzle for the same conditions of nozzle contour, mixture ratio, and vacuum thrust. Sea level thrust would be increased in the same proportions.

e. Throttled Performance

(C) Throttling performance of high pressure engines is shown in Figure 31. This figure presents the ratio of specific impulse at throttled conditions to nominal specific impulse versus percent nominal thrust as a function of mixture ratio (r = 5.0 to r = 7.0). The effect of nozzle expansion ratio on throttled performance is insignificant (less than 0.1%) and can be disregarded.



(U) Figure 31. Vacuum Specific Impulse for Throttled Conditions DFC 69984

3. Engine Weight

a. General

(U) Engine dry weight for fixed and two-position nozzle engine configurations are presented in Figure 32 through 40 for MC_8, base, and MSA nozzle contours as functions of vacuum thrust and nozzle expansion ratio.

(U) The total engine dry weight includes the weight of all engine components on the engine side of the vehicle/engine interface. Specifically, it includes the following:

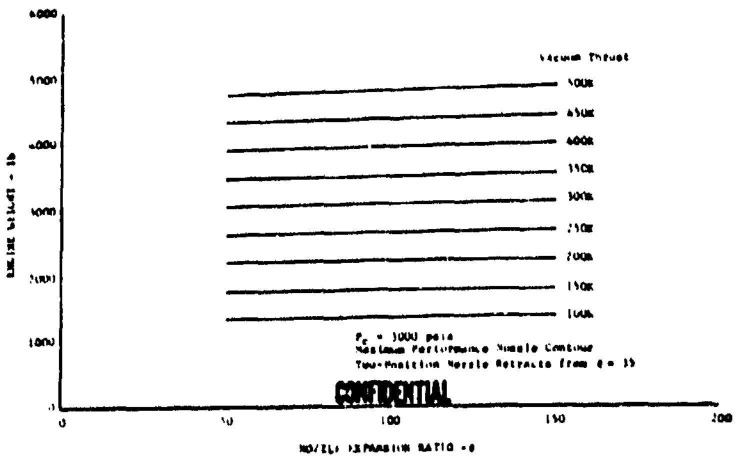
1. Thrust chamber (transpiration cooled)
2. Exhaust nozzle (regenerative and dump-cooled sections)
3. Nozzle translating mechanism (where applicable)
4. Preburner assembly
5. Fuel and oxidizer turbopump assemblies
6. Transition case and gimbal
7. Engine-mounted and driven fuel and oxidizer low-speed inducers
8. Ignition system
9. Engine controls, shutoff valves, and actuators; and plumbing
10. Gimbal and actuator arm attachment brackets for mechanical thrust vector control.

Flight instrumentation and its hardware, and TVC actuator mechanisms are not included in engine weight.

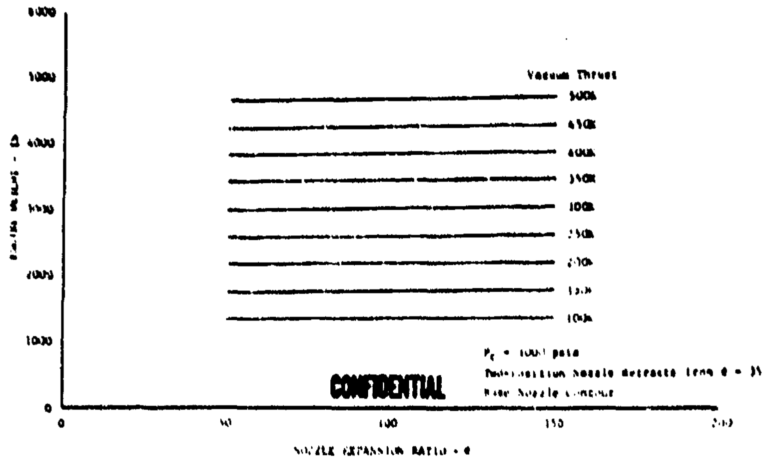
b. Two-Position Nozzle and Fixed Nozzle Engine Configurations

(C) The total weight of two-position nozzle engines having a constant primary nozzle expansion ratio, $\epsilon_p = 35$, are presented in Figures 32 through 34 for a nozzle expansion ratio range of $\epsilon = 50$ to $\epsilon = 150$. Engine weights for configurations having ϵ_p = minimum are shown in Figures 35 through 37 for nozzle expansion ratios $\epsilon = 80$ to $\epsilon = 400$. Fixed nozzle engine weights are presented in Figures 38 through 40. Engine weight differences between the two translating nozzle configurations, $\epsilon_p = 35$ and ϵ_p for minimum stowed lengths, are caused by varying nozzle translating mechanism requirements.

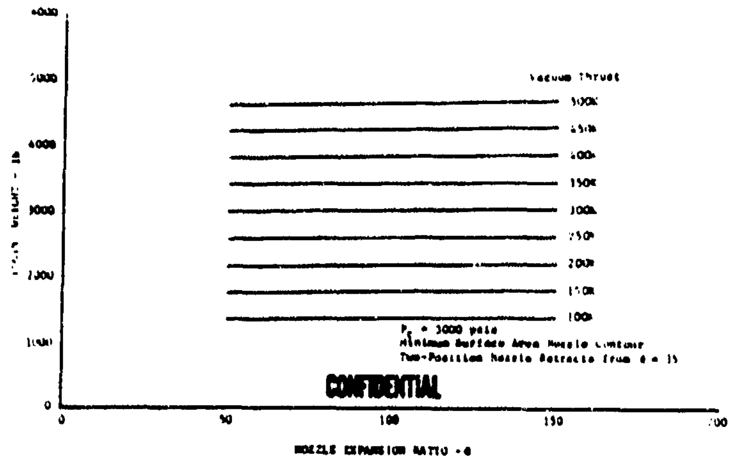
(U) Relatively flat slopes for engine weight as a function of nozzle expansion ratio are obtained with dump-cooled, lightweight nozzle engine configurations. This is a result, primarily, of using lightweight nozzle construction for dump cooling beyond an area ratio of 35. As the overall nozzle expansion ratio is increased, the surface area of the regeneratively cooled portion of the nozzle (to $\epsilon = 35$ in all cases) becomes smaller because of the change in the contouring of the nozzle (which results from the change in overall expansion ratio) and thus the nozzle becomes lighter. Conversely, the dump-cooled portion of the nozzle becomes larger and increases in weight as overall nozzle expansion ratio is increased. The net result is a relatively small increase in total engine weight with increasing expansion ratio for all thrust levels.



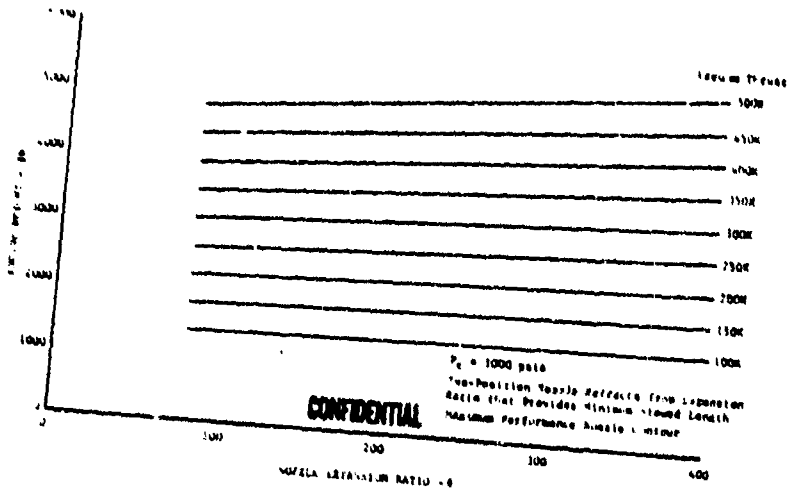
(U) Figure 32. Total Engine Weight vs Nozzle Expansion Ratio DFC 70300



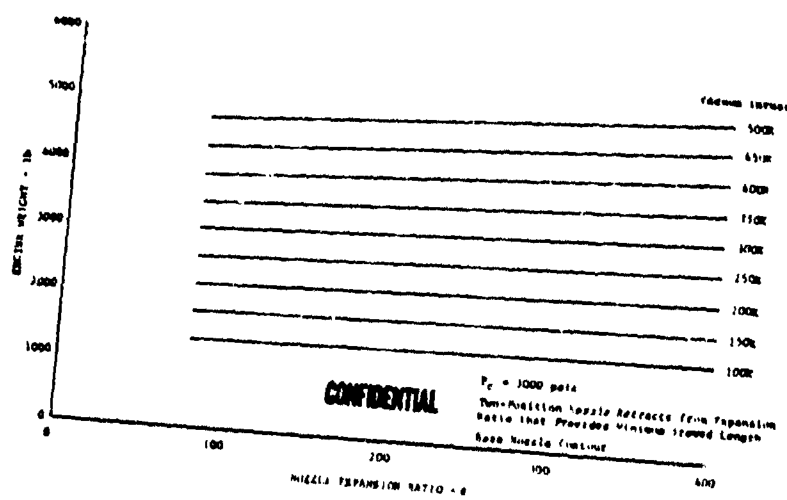
(U) Figure 33. Total Engine Weight vs Nozzle Expansion Ratio DFC 70301



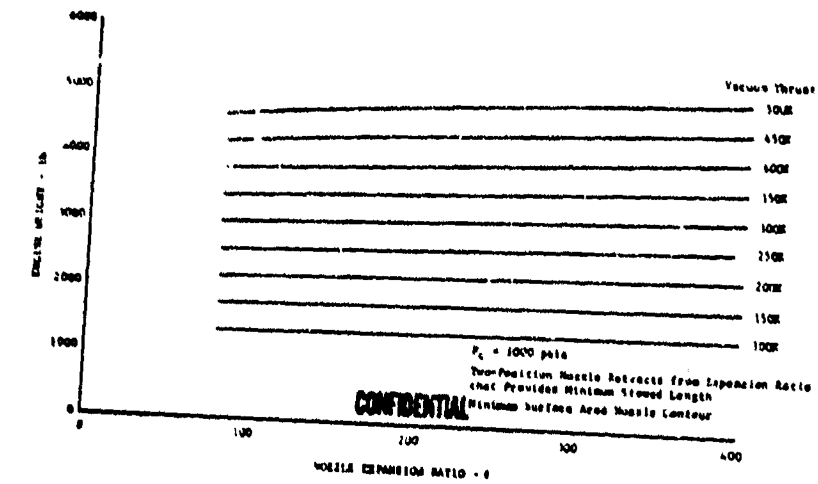
(U) Figure 34. Total Engine Weight vs Nozzle Expansion Ratio DFC 70302



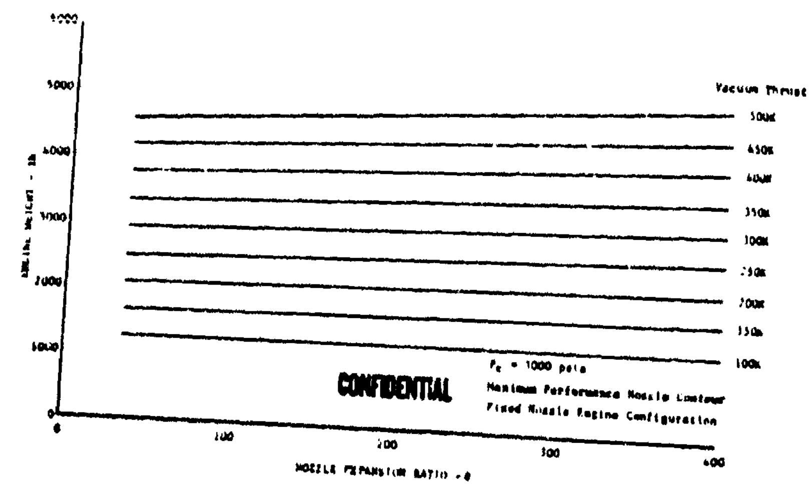
(U) Figure 35. Total Engine Weight vs Nozzle Expansion Ratio DFC 70303



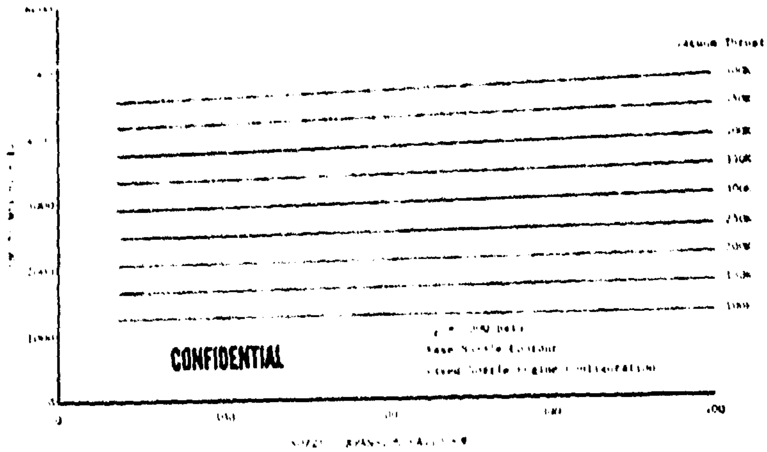
(U) Figure 36. Total Engine Weight vs Nozzle Expansion Ratio DFC 70304



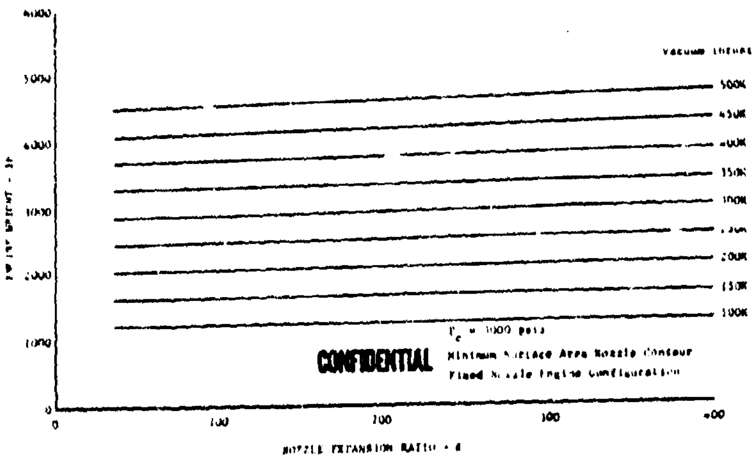
(U) Figure 37. Total Engine Weight vs Nozzle Expansion Ratio DFC 70305



(U) Figure 38. Total Engine Weight vs Nozzle Expansion Ratio DFC 70306



(U) Figure 39. Total Engine Weight vs Nozzle Expansion Ratio DFC 70307



(U) Figure 40. Total Engine Weight vs Nozzle Expansion Ratio DFC 70308

4. Engine Envelope

a. General

(U) Engine dimensions of overall (nozzle) exit diameter, overall length, and stowed lengths (for two-position nozzle engine configurations) are presented in this section in Figures 41 through 50 as a function of nozzle expansion ratio. Length data are presented for three nozzle contours; MC_s, base, and MSA. Because overall exit diameter is not significantly affected by nozzle contour, only a single figure is presented for this parameter. Primary nozzle expansion ratio as a function of overall nozzle expansion ratio for minimum stowed length engine configurations is presented in Figures 51 through 53.

(C) Stowed engine length for two-position nozzle engine configurations, in addition to being a function of thrust level and nozzle expansion ratio, is also dependent upon the expansion ratio of the primary nozzle. For lower stage applications, a small primary nozzle expansion ratio, $\epsilon_p = 35$, generally produces the best performance; in upper stages, larger primary expansion ratios, $\epsilon_p \geq 80$, provide minimum stowed length.

(C) When the primary nozzle expansion ratio is set at a constant value (i.e., $\epsilon_p = 35$), the primary nozzle exit plane determines the stowed length for low overall expansion ratios. As the overall expansion ratio is increased, a point is reached where the two-position nozzle determines the stowed length. The inflection points in the curves of stowed length occur where the exit planes of the primary and two-position nozzles are in alignment.

b. Diameter

(C) Overall exit diameter is presented in Figure 41 as a function of vacuum thrust for nozzle expansion ratios $\epsilon = 35$ to $\epsilon = 400$. This curve may be used for all nozzle truncations (contours).

c. Length

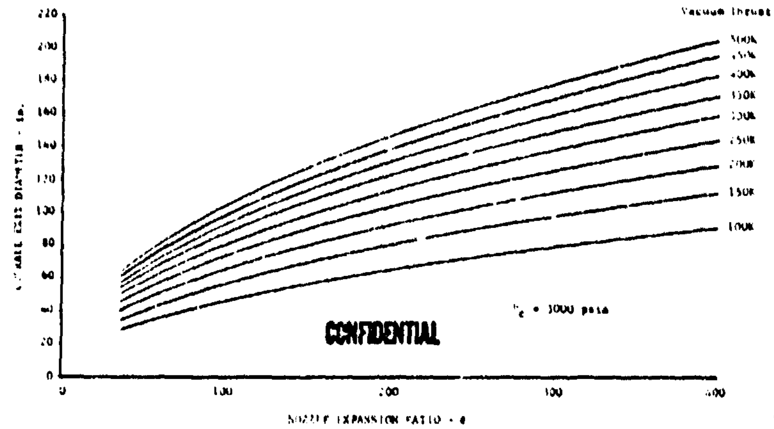
(1) Overall Length

(C) Engine overall length for fixed and two-position nozzle configurations is presented in Figures 42 through 44 for each nozzle contour as a function of nozzle expansion ratio, $\epsilon = 35$ to $\epsilon = 400$.

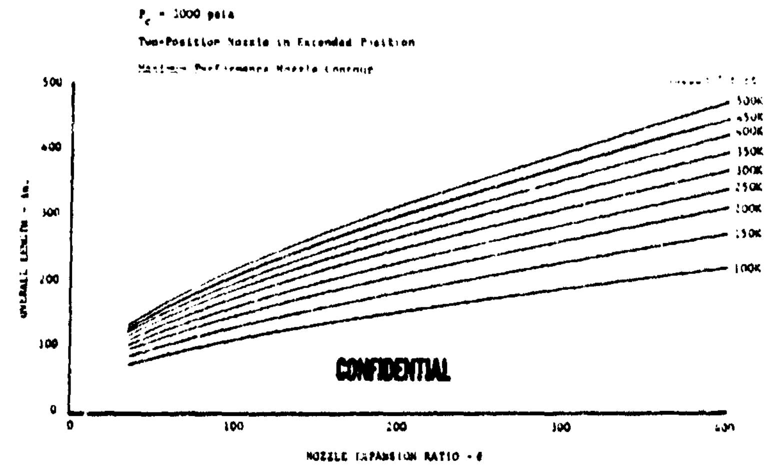
(2) Stowed and Minimum Stowed Length

(C) Engine stowed length for $\epsilon_p = 35$ is presented in Figures 45 through 47. Minimum stowed length (ϵ_p = minimum) is presented in Figures 48 through 50. Stowed length curves are presented for each of three nozzle contours as a function of nozzle expansion ratio.

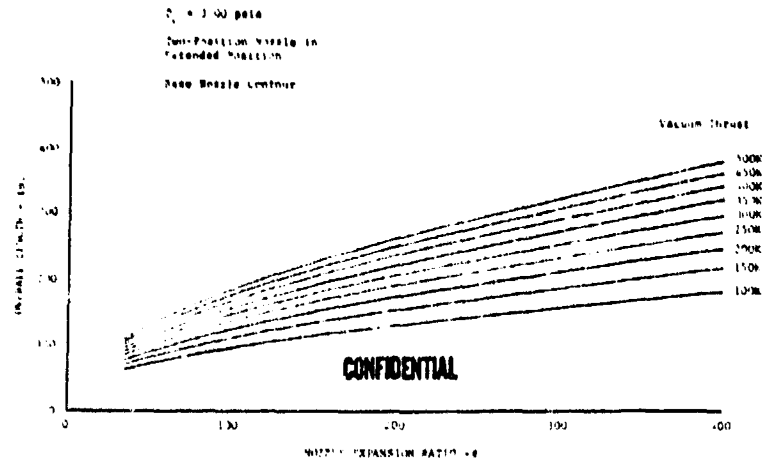
(U) For minimum stowed engine length, two hardware geometry considerations are the determining factors: (1) the nozzle translating mechanism and (2) the turbomachinery or power package. Inflections in the minimum stowed length curves are caused by a changeover in the limiting factor.



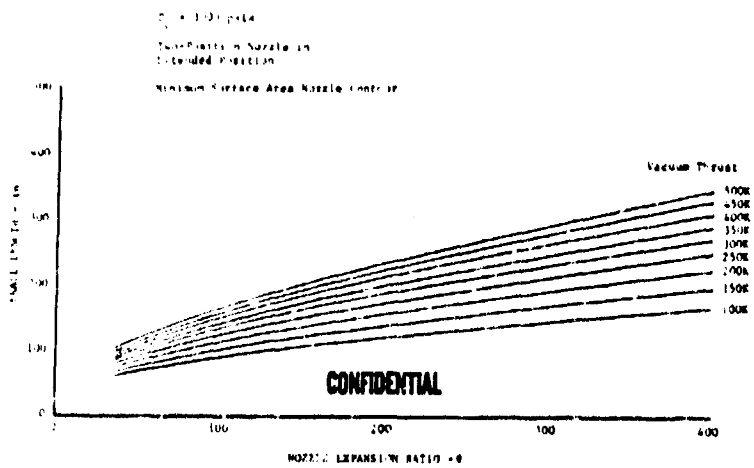
(U) Figure 41. Overall Exit Diameter vs Nozzle Expansion Ratio DFC 70274



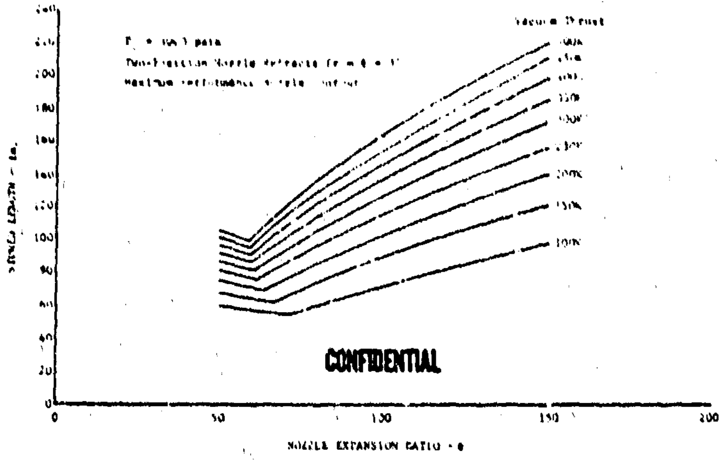
(U) Figure 42. Overall Length vs Nozzle Expansion Ratio DFC 70275



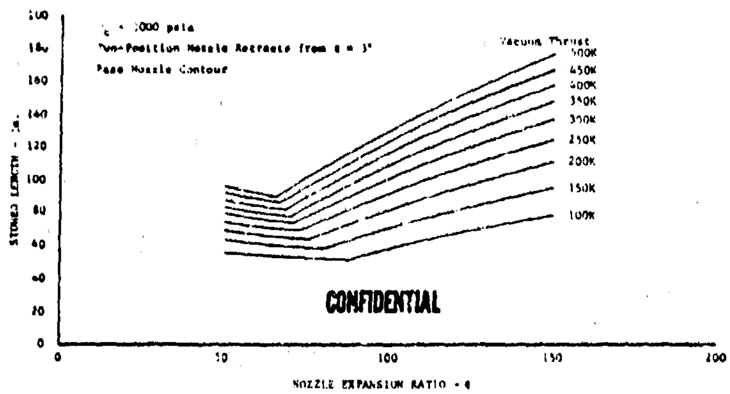
(U) Figure 43. Overall Length vs Nozzle Expansion Ratio DFC 70276



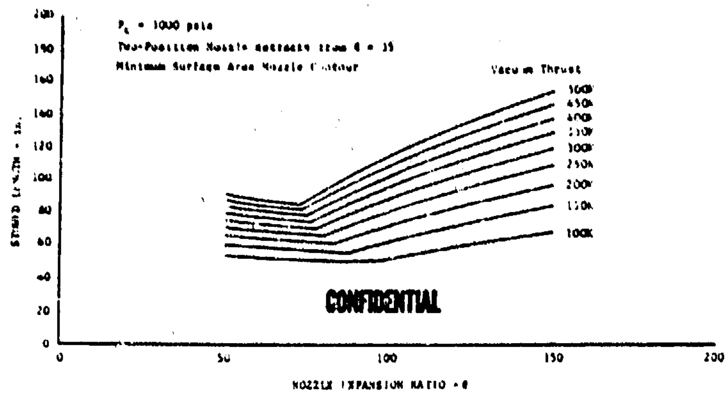
(U) Figure 44. Overall Length vs Nozzle Expansion Ratio DFC 70277



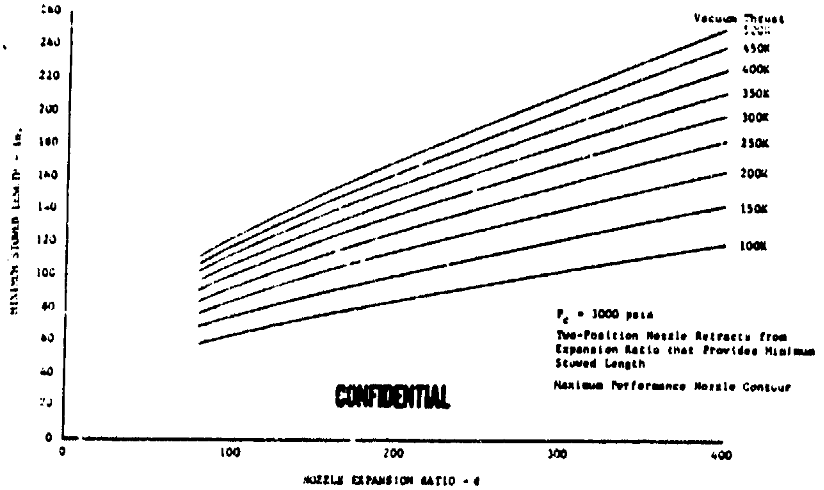
(U) Figure 45. Stowed Length vs Nozzle Expansion Ratio DFC 70366



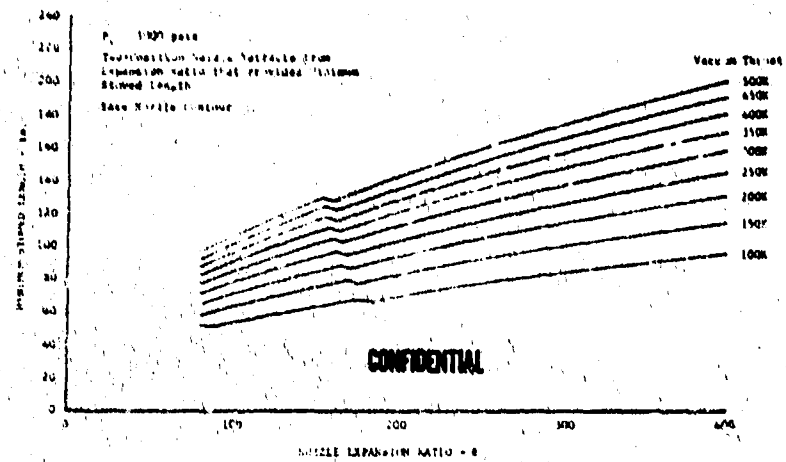
(U) Figure 46. Stowed Length vs Nozzle Expansion Ratio DFC 70365



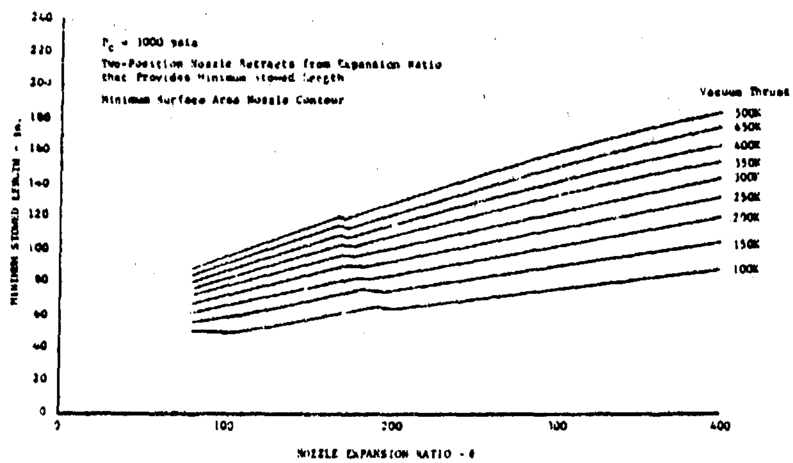
(U) Figure 47. Stowed Length vs Nozzle Expansion Ratio DFC 70364



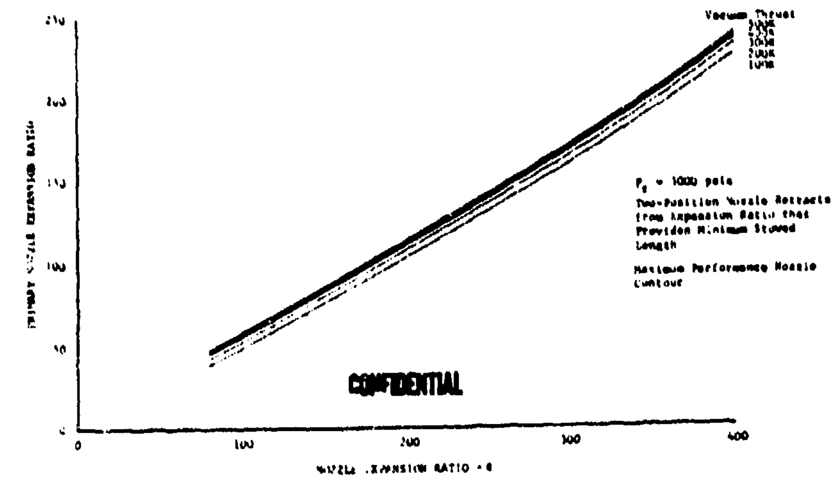
(U) Figure 48. Minimum Stowed Length vs Nozzle Expansion Ratio DFC 70363



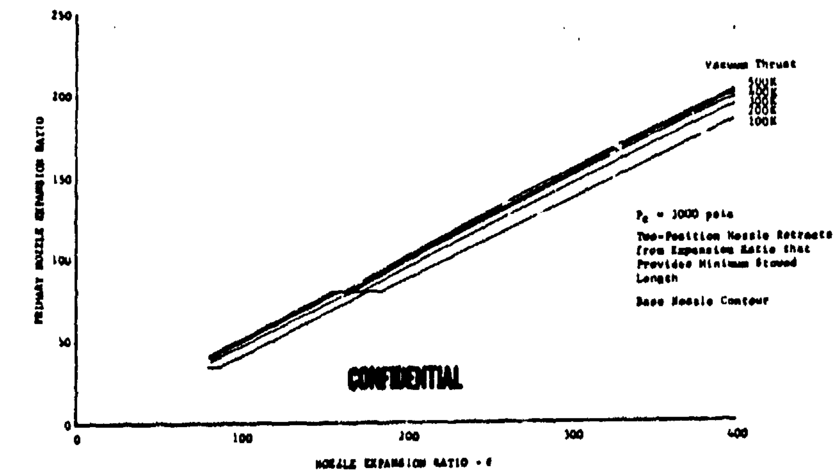
(U) Figure 49. Minimum Stowed Length vs Nozzle Expansion Ratio DFC 70362



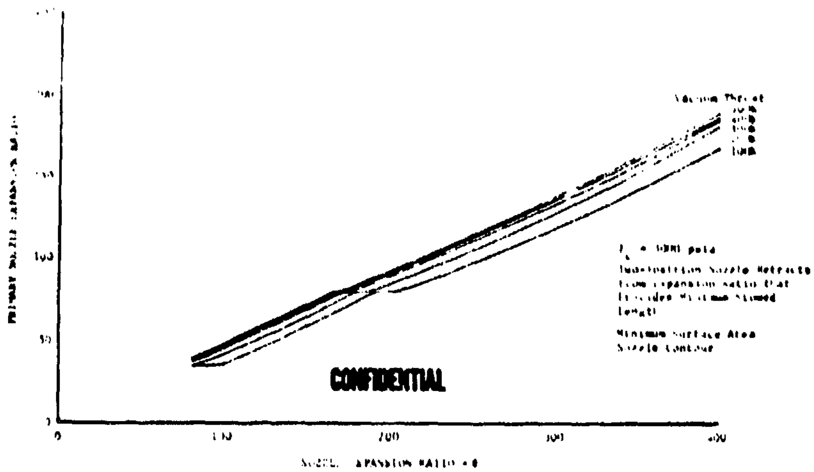
(U) Figure 50. Minimum Stowed Length vs Nozzle Expansion Ratio DFC 70361



(U) Figure 51. Primary Nozzle Expansion Ratio vs Nozzle Expansion Ratio DFC 70360



(U) Figure 52. Primary Nozzle Expansion Ratio vs Nozzle Expansion Ratio DFC 70358



(U) Figure 53. Primary Nozzle Expansion Ratio vs Nozzle Expansion Ratio DFC 70359

D. ENGINE/VEHICLE INTERFACE DATA

1. General

(U) Engine/Vehicle interface data are presented in the tables of this section as a function of thrust level in 50K increments. These data are based on the demonstrator engine configuration. These interface data include inlet condition operating region, engine inlet and TVC actuator arm attach point locations, engine inlet and power package diameters, TVC actuator arm loads and auxiliary power available from the engine.

2. Engine Inlet Conditions

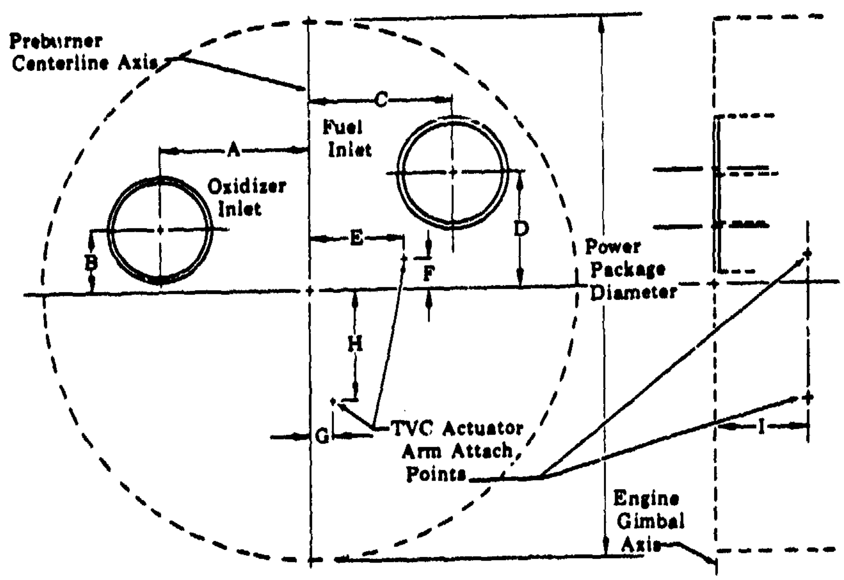
(C) The high pressure engines are designed to be capable of operating over a wide range of fuel and oxidizer inlet conditions provided minimum net positive suction head (NPSH) requirements are met. The flight engine inlet operating regions are the same as those for the demonstrator engine (see Figures 5 and 6) with minimum required NPSH's of 60 feet and 16 feet at the fuel and oxidizer inlets, respectively. If special vehicle operating conditions require inlet conditions outside of the normal engine operating regions, these should be coordinated with Pratt & Whitney Aircraft to ensure engine/vehicle compatibility.

3. Engine/Vehicle Interface Locations

(U) Engine inlet and actuator arm attach point locations are presented in Table VI. These dimensions are referenced to the engine X, Y, and Z axes shown in Figure 54. The engine fuel and oxidizer inlet flanges are in the same plane as the gimbal axis of the engine.

(U) Table VI. Location of Engine Inlets and Actuator Arm Attach Points[1]

	Vacuum Thrust (Thousands of lb)								
Dimension (in.)	100	150	200	250	300	350	400	450	500
A	12	14	16	18	20	22	23	25	26
B	5	6	7	8	8	9	10	10	11
C	11	14	16	18	19	21	22	24	25
D	9	11	13	14	15	17	18	19	20
E	8	9	11	12	13	14	15	16	17
F	3	3	4	4	4	5	5	5	6
G	2	2	2	3	3	3	3	4	4
H	8	10	12	13	14	15	16	17	18
I	20	20	21	21	22	22	23	23	24



(U) Figure 54. Engine/Vehicle Interface Locations

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[1]Refer to Figure 54

4. Inlet Line and Power Package Diameters

(U) Fuel and oxidizer inlet line diameters at the engine/vehicle interface (low-speed inducer inlets) are presented in Table VII. The diameters of the inlets are determined by propellant NPSH and engine cycle requirements.

(U) The engine power package consists of the turbomachinery, preburner, main burner injector and chamber and the associated plumbing lines; it is essentially the entire engine except for the exhaust nozzle. The power package maximum diameter occurs in the plane of the main turbopumps and preburner parallel to the gimbal axis, and generally does not exceed a diameter equivalent to that for a nozzle expansion ratio of 80. The power package diameters are presented in Table VII.

(U) Table VII. Power Package Diameter, Oxidizer and Fuel Inlet Diameters

	Vacuum Thrust (Thousands of lb)								
	100	150	200	250	300	350	400	450	500
Power Package Diameter (in.)	41	50	58	65	71	76	82	87	92
Oxidizer Inlet Diameter (in.)	7	9	10	12	13	14	15	16	17
Fuel Inlet Diameter (in.)	8	9	11	12	13	15	16	17	18

5. Gimbal Loads

(C) Thrust vector control (TVC) for the high pressure engines is accomplished by mechanical gimbaling. Two actuator arms attach to the engine at points 90 degrees apart. The estimated maximum gimbal loads (for each actuator arm) are presented in Table VIII. Gimbal loads were based on the following gimbaling requirements:

Angle - ± 7 deg

Velocity - 30 deg/sec

Rate - 30 rad/sec^2

6. Auxiliary Power

(U) The high pressure engines are designed to provide auxiliary (accessory) power for TVC and other uses. The maximum power availability is presented in Table VIII.

(U) Table VIII. Gimbal Loads and Auxiliary Power Available from the Engine

	Vacuum Thrust (Thousands of lb)								
	100	150	200	250	300	350	400	450	500
Gimbal Loads (Thousands of lb)	16	22	28	33	39	45	51	56	62
Auxiliary Power (Horsepower)	43	60	80	100	117	134	151	168	185

REFERENCES

*1. High Pressure Oxygen-Hydrogen Rocket Engine Parametric Design Data, PWA FR-492D, 30 November 1966

*2. Applications Study for a High Performance Cryogenic Staged Combustion Rocket Engine, Final Report, AFRPL-TR-67-270, November 1967

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13. ABSTRACT

This special technical report presents information and data on the XLR129-P-1 rocket engine. Information is presented for both the demonstrator engine and flight engine versions of this rocket engine. A general description and pertinent technical information are presented for the demonstrator engine. The demonstrator engine program schedule is also presented. Parametric design, performance, cost, and schedule data are presented for the flight engine. This technical report has been prepared for the use of airframe manufacturers and government personnel who are conducting mission and vehicle studies.

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Pratt & Whitney Aircraft, FRDC						
XLR129-P-1 Demonstrator Engine						
Installation Drawing						
Flow Schematic						
Operating Parameters						
Flight Engine Description						
Nozzle Concepts						
Parametric Engine Data						
Engine Performance						
Engine Weight						
Envelope						
Engine/Vehicle Interface Data						

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